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Development of a field repair technique for mini-sandwich Kevlar/epoxy aircraft skin.

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NAVAL POSTGRADUATE SCHOOL

Monterey, California

THESIS

DEVELOPMENT OF A FIELD REPAIR TECHNIQUE
FOR "MINI-SANDWICH" KEVLAR/EPOXY
AIRCRAFT SKIN

by

David Bruce Cripps

June 1984

Thesis Advisor: R. L. Foye

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Development of a Field Repair Technique for "Mini-Sandwich" Kevlar/Epoxy Aircraft Skin

by

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Captain, United States Army
B.S., United States Military Academy, 1978

Submitted in partial fulfillment of the requirements for the degree of

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from the

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June 1984

ABSTRACT

An experimental analysis was performed on Kevlar/epoxy cloth "mini-sandwich" panels with cellular foam core. Three undamaged panels and twenty-three other panels with damage and repairs were subjected to static shear loading. Four parameters were varied in the types of repairs; hole size, hole plug filler material, patch material, and patch overlap distance. All twenty-six panels were tested to failure. A repair technique employing a cellular foam plug and fiberglass patches overlapping the original hole by 0.50 inches, symmetrically applied with structural adhesive, was found to be suitable for repair of up to three inch diameter circular holes at field repair level. Additionally, postbuckling energy absorption was qualitatively examined for undamaged panels and for hole sizes ranging from 1.00 to 5.00 inches diameter.

TABLE OF CONTENTS

LIST OF FIGURES

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I. INTRODUCTION

The construction of metal semi-monocoque structures has dominated the aviation industry for the past half century. In these structures, internal components such as stringers, ribs, longerons and spars generally carry the flight loads and bending moments, and the skin (as well as some internal walls and bulkheads) carry the shear loads. The most widely used skin material has been aluminum, largely due to its light weight and favorable mechanical properties.

The use of advanced composite materials in aviation applications has been a rather recent phenomenon. Initially employed to achieve weight savings, composites have been used to fabricate fairings and other aerodynamic surfaces which carried little actual load. Within the last decade, composites have gained increased usage in structural components, primarily internal bulkheads and panels. Attempts to make large scale use of advanced composites on fuselage skins have been periodically attempted, and for the most part have been discontinued due to difficulties which arose. Among these difficulties have been complications in fastening adjoining components, lack of out of plane rigidity of thin skins, and expense.

The development of lightweight advanced composite skin designs has been hampered primarily by the lack of out of plane rigidity. Advanced composite materials sized to carry the same loads as conventional aluminum skins are on the average only a few hundredths of an inch thick. This gives rise to their weight savings of approximately twenty-five percent over conventional aluminum skins. The composite skins are generally woven fabrics with epoxy filler, or matrix, penetrating and encasing them. Due to their thinness, they exhibit little resistance to out of plane loads, resulting in deformation in the form of wrinkling or other buckling phenomena. To increase the rigidity of the skins to out of plane loads, several schemes for increasing the out of plane moment of inertia have been developed. Among these are "sandwich" type designs, in which composite fabric skins are laid up above and below a core material. This core material has been primarily honeycomb Nomex or honeycomb aluminum. Cellular foams initially were incapable of sustaining the pressures and temperatures necessary to cure the resins in the composite fabrics. Recent developments in foams have resulted in the production of a cellular foam which is compatible with the fabrication process for composite fabrics. This has enabled the development of a "mini-sandwich" skin design, in which a foam core on the order of a qurater of an inch thick is laid up between single sheets of +/- 45 degree weave Kevlar/epoxy preimpregnated (prepreg) fabric. When co-cured, the components adhere to form a unitized, lightweight skin with increased out of plane rigidity.

Many problems inherent in metal skin designs are resolved by use of the "mini-sandwich" design. Vast reductions in the number of mechanical fasteners required to maintain structural integrity are experienced. Complex contours can be readily shaped in the fabrication process, as all components of the skin can be easily formed prior to curing. Weight savings on the order of twenty-five percent can be achieved over aluminum skins. Adverse environmental effects such as corrosion are greatly reduced.

Inadvertant damage to aircraft skin necessitates a methodology for repair. Traditional mechanically fastened patches are inappropriate for "mini-sandwich" skins. Such repairs create additional sites for stress concentration and fail to take advantage of the favorable properties of advanced composite materials. A repair technique tailored to the nature of the material at hand is required.

Catastrophic damage to the skin would likely result in destruction of a major component or loss of the aircraft. Incidental damage, however, may not initially degrade the structural integrity of the airframe. The repair of such incidental damage is most appropriately accomplished at the user level, where loss of utilization of the aircraft during the time of repair can be kept to a minimum. Accordingly, this investigation was conducted to develop a field repair technique for Kevlar/epoxy "mini-sandwich" aircraft skin material.

II. METHOD OF INVESTIGATION

In order to select an effective field repair technique, a variety of possible methods were investigated. Inherent in each method was the assumption that the damaged area could be prepared for the repair by cutting a smooth circular hole entirely through both inner and outer skin surfaces, the foam core material removed and replaced, and equally sized and shaped patches applied to both surfaces (i.e no asymmetric patches). The diameter of the hole would be equal to the longest cross section distance of the original damaged area. This method of smoothing to a clean circular hole minimizes stress concentrations otherwise inherent in the small radii of curvature present in the rough flaw surfaces.

Four parameters were selected as ones which would provide meaningful results when varied; hole size, patch overlap distance, patch material, and plug filler material. Twenty-six panels were investigated by varying these four parameters. Table 1 lists the panels and their respective configurations. Of the twenty-six panels, three had neither holes nor repairs, and provided baseline data against which the other panels could be compared.

The entire stock of panels was manufactured in a single large sheet by co-curing Kevlar/epoxy inner and outer skin surfaces to the foam core material. Individual panels were

TABLE 1

CONFIGURATION OF TEST PANELS

Panel	Hole Size (in.)	Plug Material	Patch Material	Patch Radius Overlap (in.)
A	none	none	none	none
B	3.00	foam	Kevlar	0.25
C	3.00	foam	Kevlar	0.50
D	3.00	foam	Kevlar	0.75
E	2.00	foam	Kevlar	0.25
F	2.00	foam	Kevlar	0.50
G	2.00	foam	Kevlar	0.75
H	2.00	foam	fiberglass	0.25
I	2.00	foam	fiberglass	0.50
J	2.00	foam	fiberglass	0.75
K	2.00	adhesive	Kevlar	0.25
L	2.00	adhesive	Kevlar	0.50
M	2.00	adhesive	Kevlar	0.75
N	2.00	none	Kevlar	0.25
O	2.00	none	Kevlar	0.50
P	2.00	none	Kevlar	0.75
Q	0.50	foam	Kevlar	0.25
R	0.50	foam	Kevlar	0.50
S	0.50	none	none	none
T	2.00	none	none	none
U	1.00	none	none	none
V	3.00	none	none	none
W	4.00	none	none	none
X	5.00	none	none	none
Y	none	none	none	none
Z	none	none	none	none

then cut from the large sheet. Each panel was visually inspected for flaws and delaminations. In accordance with Table 1, holes were cut in the panels and repaired. Several panels were tested with no repairs over the holes. The panels were instrumented with bonded strain gages, and mounted in a "picture frame" shear fixture. A tensile load was applied to two opposite corners of the fixture, and the panels were loaded to failure. For the purposes of this investigation, failure was defined as macroscopic irreversible deformation or damage to the panel. Strain gage output was recorded at regular load intervals throughout the entire loading process. Additionally, several panels were loaded beyond the initial point of failure, to the point of collapse. Strain gage data was taken during that process as well. Test results were then evaluated, and conclusions were drawn.

III. PANEL DESIGN

A. PANEL LAYUP

The "mini-sandwich" skin design examined in this investigation is that employed by Bell Helicopter Textron for portions of the fuselage skin on their Advanced Composite Airframe Program (ACAP) testbed aircraft. Fabrication of the panels was conducted by Bell Helicopter Textron, while fitting of the panels to the test fixture and all repairs were accomplished locally.

The panels were constructed of 281 weave prepreg K-49/CE306, a woven Kevlar/epoxy system produced by Ferro Corporation, and of Rohacell 71WF, a closed-cell rigid polyimide foam produced by Cyro Industries. The desired panel was to be a thin, symmetric layup which would provide significant shear strength, yet be very lightweight. The resultant layup was a layer of +/- 45 degree Kevlar/epoxy prepreg fabric sandwiched above and below a core of 0.110 inch thick Rohacell 71WF foam. After co-curing, the skin material had an approximate thickness of 0.155 inches. The weight of this combined layup was 0.480 pounds per square foot. This represents a savings of nearly 33 percent over 0.050 inch thick 2024-T4 aluminum, and 17 percent over 0.040 inch thick aluminum.

The 281 weave prepreg K-49/CE306 Kevlar/epoxy fabric is a 250 degree (Fahrenheit) curing system. Averaging the shear properties of the 250 degree cure Kevlar/epoxy systems provided by DuPont (manufacturers of Kevlar fiber), resulted in an interlaminate shear strength of 7000 psi at room temperature, and an in plane shear modulus of 3,000,000 psi. Rohacell 71WF has a shear strength of 185 psi at room temperature, and an average shear modulus of 4402 psi. The average in plane shear modulus of the "mini-sandwich" layup was determined in the following manner.

$$\overline{G} = \{(G_1 t_1) + (G_2 t_2)\} / (t_1 + t_2)$$

This gave an average in plane shear modulus of 803,000 psi.

B. DESIGN CRITERIA

After comparing fifteen load cases for the ACAP aircraft by use of MSC/NASTRAN finite element computer analysis, Bell Helicopter Textron defined a limit critical load case as a 20 foot per second landing with 10 degree pitch and 10 degrees roll, resulting in a limit shear flow of 37 pounds per inch. Additionally, an ultimate critical load case was defined as a 42 foot per second vertical level crash, resulting in an ultimate shear flow of 342 pounds per inch. The "mini-sandwich" is desired to be linearly shear resistant at limit load and diagonally tension safe at ultimate load. The desired limit shear strain is then calculated from the limit shear flow.

$$\gamma_{lim} = \tau_{lim} / \bar{G} = q_{lim} / (\bar{G}\ t) = 0.000307 \text{ in/in}$$

Likewise, the ultimate shear strain is estimated from the ultimate shear flow.

$$\gamma_{ult} = \tau_{ult} / \bar{G} = q_{ult} / (\bar{G}\ t) = 0.002839 \text{ in/in}$$

C. SELECTION OF REPAIR METHODS

Each of the four parameters selected was varied within an appropriate range with respect to the expected ability of a user level maintenance activity to successfully accomplish the repair action in a timely manner. Compatibility of the repair to the surrounding skin was also considered.

Hole sizes were varied fron 0.50 inches to 5.00 inches, with repairs being applied to holes of 0.50 inches, 2.00 inches, and 3.00 inches. The larger holes were examined for their effect on shear characteristics and post-buckling behavior without repairs.

Two types of patch material were selected. Prepreg 281 weave K-49/CE306 Kevlar/epoxy fabric (250 degree cure) skin patches were chosen due to their commonality with the original skin material. Woven fiberglass fabric was also employed, due to its availability at user level maintenance activities, its ease of use, and its ability to be easily contoured upon application to non-planar surfaces. The fiberglass fabric utilized was part of a fiberglass repair kit currently in the military supply system (P/N 516077021, Parts Kit, Repair of Reinforced Fiberglass). The fiberglass fabric was adhered to the panels with EA-956 structural adhesive, manufactured by the Hysol Division of the Dexter Corporation. This adhesive was selected due to its compatibility with CE306 epoxy, fiberglass, Kevlar, and Rohacell foam.

The void created by removal of the Kevlar/epoxy and foam material was varied in three manners. The primary method was replacement of the core material with a disk shaped plug of Rohacell 71WF foam material. The second method employed was to pour structural adhesive into the void and allow it to cure there as a plug. The third variation was to allow the void to remain vacant, and merely apply a skin patch on both surfaces.

Patch radius overlap was varied from 0.25 inches to 0.75 inches for 2.00 inch and 3.00 inch diameter holes, and from 0.25 inches to 0.50 inches for 0.50 inch diameter holes.

IV. EXPERIMENTAL APPARATUS AND PROCEDURES

A. PANELS

A lot of thirty panels, measuring 12 inches square, and comforming to the layup described previously, was produced by Bell Helicopter Textron. Four panels were utilized for testing methods of drilling or cutting holes, compatibility of different adhesives, methods of affixing patches to the surfaces, alternative adhesives for application of bonded strain gages, and for assuring an adequate method for holding each panel within the test fixture. The remaining panels were configured for testing in the "picture frame" shear fixture in the manner shown in Figure 1. The four corners of each panel were removed so that the contribution of resistance to shear otherwise afforded by the corner material would be eliminated. It is important to note that the four pivot points of the test fixture (the centers of the four corner bolts) actually occurred over panel material and not over the cutout corner sections of the panels. The foam core material along each of the four sides of the panels was removed to a depth of 1.25 inches, and was replaced with aluminum spacers having a thickness of 0.110 inches. The aluminum spacers allowed sufficient torque to be applied to the nine through bolts along each side of the panel to assure negligible slippage in the frame during loading without crushing of the core material within the test area of the panel.

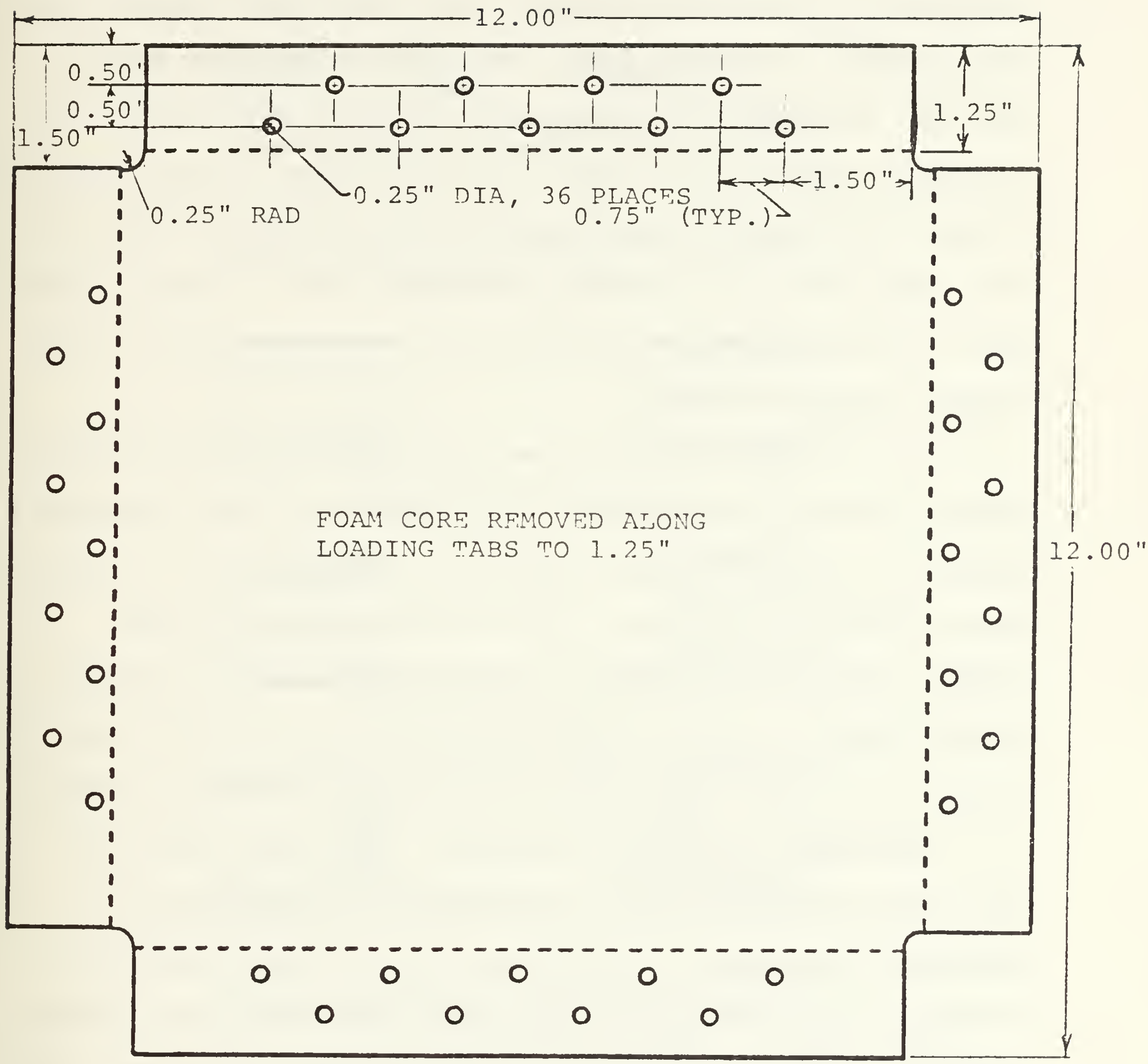


Fig. 1. Panel Specifications

Each panel requiring a hole in the test area had a pilot hole drilled through the center of the panel. Holes of 0.50 inches and 1.00 inches diameter were bored with circular hole saws mounted in a drill press. All larger diameter holes were cut by means of an adjustable circular hole cutter with a hardened steel cutting edge. For each panel a hole was cut through the top Kevlar/epoxy skin, the panel was turned over, and a hole was cut in the the other skin and through the foam core. This resulted in a smooth cut free of any significant fraying along the edge.

For each case in which there was to be a core filler to replace the cut out material, the foam core around the entire perimeter of the hole was removed to a depth of 0.125 inches from the edge of the hole. This allowed slightly more surface area for the adhesive holding the plug in place to attach itself. Figure 2 depicts how the core material was recessed for both foam plugs and structural adhesive plugs.

The method for placing Rohacell 71WF plugs into the holes was straightforward. A circular disk matching the diameter of the hole in the panel was cut from a sheet of 0.276 inch thick Rohacell 71WF foam. Its edges were lightly sanded to insure smoothness, and it was tightly fitted to the hole. The disk was then removed and cleaned by means of an air jet. EA-956 structural adhesive was applied to the inner edge of the hole where the core had been recessed. The disk was then placed in the hole, and centered in the thickness

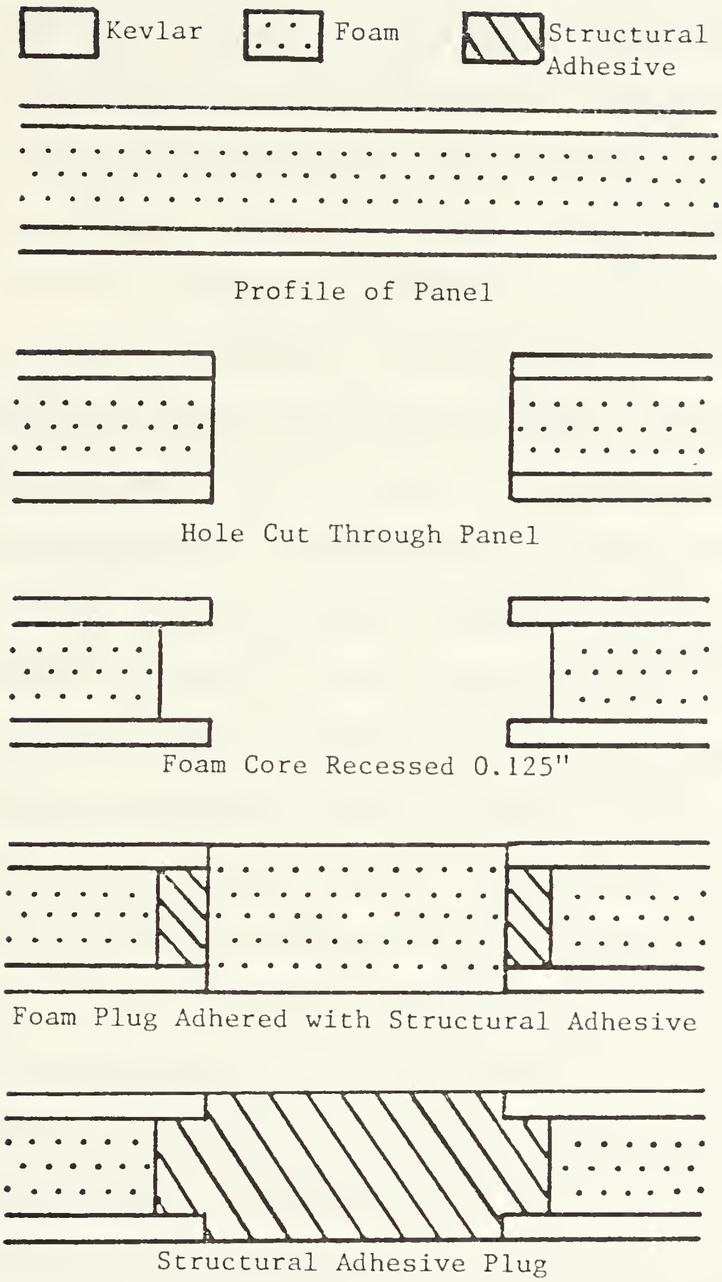


Fig. 2. Recessing of Foam Along Inner Edge of Holes

direction. The panel was then placed in an oven at 200 degrees Fahrenheit for ten minutes, during which time the adhesive hardened. Hardening of the adhesive can be accomplished at room temperature in about two hours, or in about ten minutes by employing a simple hand held heating gun. The foam plug was then sanded smooth to the level of the surrounding skin on both sides and cleaned.

Panels in which structural adhesive plugs were employed were first clamped securely to smooth aluminum sheets. EA-956 structural adhesive was poured into the hole area until the hole was filled even with the upper surface. The panel was then placed in an oven at 200 degrees Fahrenheit for one hour, during which time the plug hardened. Sample plugs of 0.25 inch thickness and 1.00 inch diameter took approximately eight hours to harden sufficiently at room temperature to be handled. After hardening, the panels were removed from the aluminum sheets and the plugs were sanded smooth to the level of the surrounding skin on both sides.

Kevlar/epoxy patches were applied in the following manner. Circular patches were cut from sheets of prepreg K-49/CE306 Kevlar/epoxy fabric. Panels were preheated to approximately 200 degrees Fahrenheit. Insuring alignment with the weave direction of the panel, the patches were carefully placed onto both sides of the panels, centered over the plug. By preheating the panels, the Kevlar/epoxy patches would soften somewhat and become tacky, allowing them to be moved a small amount once placed on the panels to insure correct positioning, and holding the patches in place while transferring the panels to the heated press for curing. The press used was a Wabash 10,000 pound capacity hydraulic press with electrically heated platens. Figure 3 shows the hydraulic press. The temperature of the platens was maintained by a Leeds and Northrup Speedomax H temperature controller unit in conjunction with a locally fabricated timer unit, shown in Figure 4. Panel temperature was sensed by a thermocouple placed on the panel and maintained at 250 degrees Fahrenheit by the controller. Approximately 45 psi of pressure was applied to the patches by means of the hydraulic cylinder in the press. Temperature and pressure were maintained for one hour, at which time the panels were removed from the press and inspected. Those panels in which no filler plug was employed had the skin patches applied one at a time. In these cases, the side to which the patch was being applied was placed downward in the press. As it cured at temperature and pressure, gravity would hold the skin under the hole against the lower platen. When that side was cured, the remaining patch was applied to the other side, which in turn was placed downward in the press. Since the first patch had already cured, it would not sag into the lower patch and allow them to fuse together.

The panels which utilized fiberglass patches were preheated to 200 degrees Fahrenheit. A thin layer of EA-956

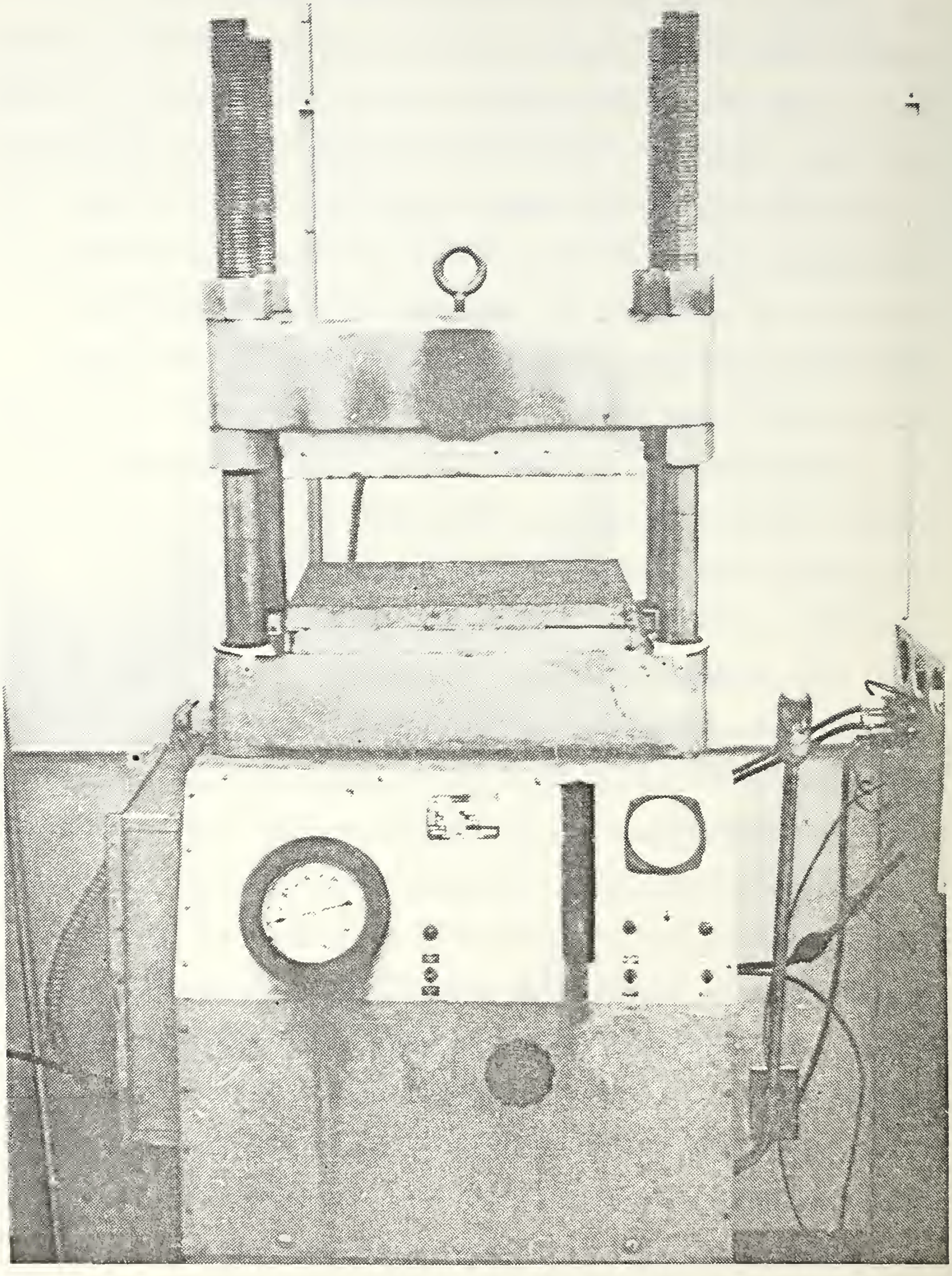


Fig. 3. Wabash Hydraulic Press

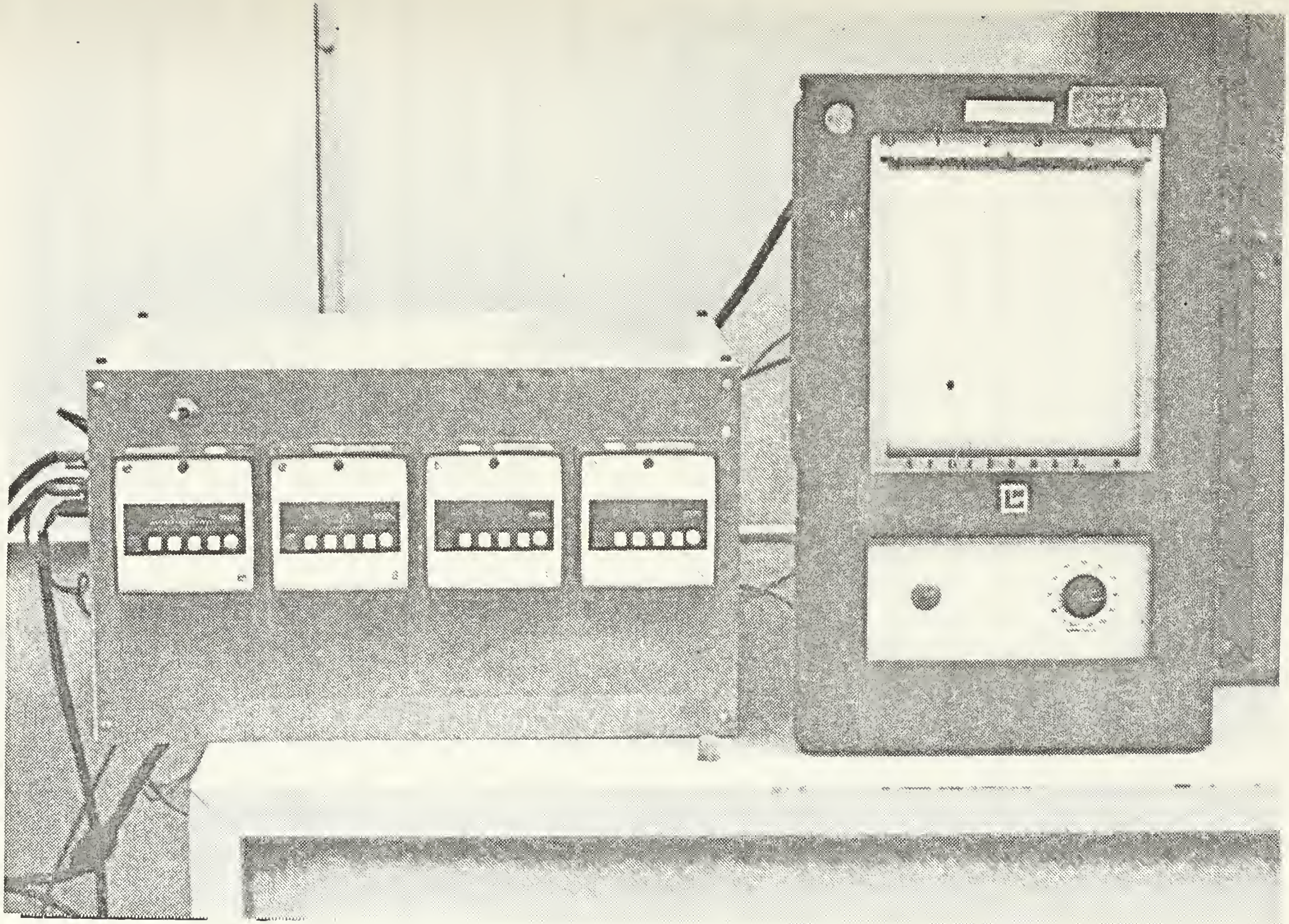


Fig. 4. Speedomax H and Timer Unit

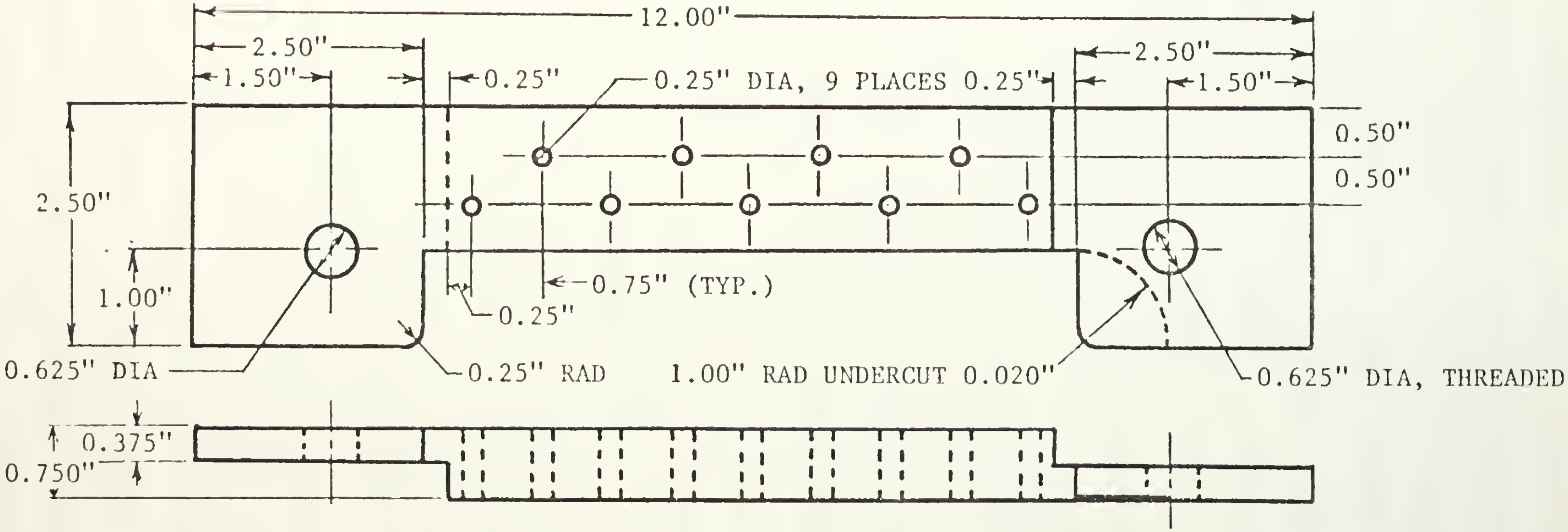
structural adhesive was applied to the plug surface and around the edge of the hole on one side of the panel. A circular patch of woven fiberglass material was then placed onto the panel and centered over the plug. The fiberglass was tamped lightly into the adhesive until it was saturated. Additional adhesive was applied to insure complete saturation of the fiberglass and smoothness of the patch surface. The panel was then placed into an oven at 200 degrees Fahrenheit for ten minutes, during which time the adhesive hardened. The panel was then inverted, and the procedure was repeated on the other side.

Once the repairs had been accomplished, all repairs were allowed to cure additionally at room temperature for twenty-four additional hours prior to testing.

B. TEST FIXTURE

The disposition of "picture frame" shear fixtures to pinch the panels at the tension corners where load is applied (causing a scissors-like action on the panel) and to separate loading tabs at the unloaded corners (causing excessive stretching in the panel) can be overcome by designing the fixture such that its corner pins coincide with the corners of the test area of the panel [Ref. 1]. To accomplish this, a shear fixture was designed taking into account colocation of the corners of the panel's test area and the "picture frame" corner pins. The corner pins were designed to be separate on front and back halves of the frame, rather than penetrate the test panel, allowing part of the panel to lie directly below the corner pins of the fixture. This allows the pivot point of the test fixture to occur at the corner of the panel test section without drilling a pivot pin hole in the test panel, which would induce a stress concentration at the corner. Figure 5 illustrates the specifications to which the components of the frame were fabricated. The assembled frame is shown in Figure 6. The frame was machined from stainless steel, with each piece being nominally 0.50 inches thick , 1.50 inches wide and 12.00 inches long. The frame was designed to be sufficiently massive such that an assumption of frame rigidity would be reasonable. All twenty-six panels were placed into the fixture in the same orientation, and secured with thirty-six 1/4-inch bolts, each of which was torqued to 125 inch-pounds.

In order to insure that the state of stress developed within the shear fixture was uniform over a large portion of the test area, a test panel composed solely of 0.125 inch thick sheet photoelastic material was prepared and secured in the "picture frame" with 30 inch-pounds of torque applied to the thirty-six through bolts. The photoelastic panel had been prepared by coating one side evenly with a reflective aluminized paint. A polariscope and camera arrangement was used to photograph the resulting stress distribution as exhibited by the photoelastic process. Appendix A contains



This drawing depicts a front frame member. Rear frame members are mirror images of this member, with the nine 0.25" holes threaded.

Fig. 5. Drawing of Shear Fixture Member

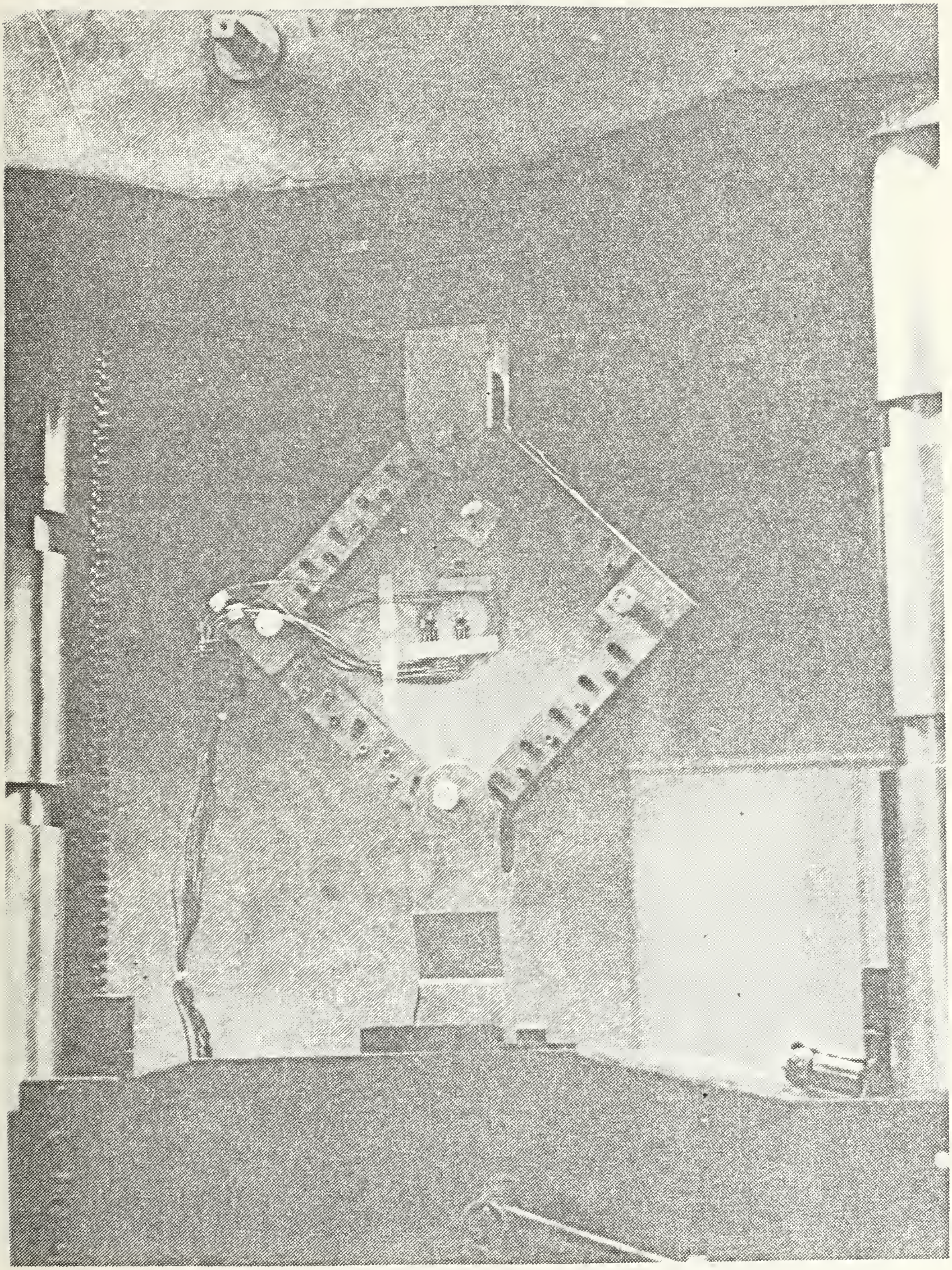


Fig. 6. Sample Panel in Shear Fixture

photographic results of the stress distribution throughout the load cycle. It can be seen from these results that uniformity of stress in fact did exist across nearly the entire test area at higher loads, with the stress gradients being restricted to the regions immediately surrounding the corners and near the edges.

C. EXPERIMENTAL SETUP

Each panel was instrumented with three bonded strain gage rectangular rosettes and a linear bonded strain gage. The rosettes utilized were EA-06-060RZ-120 rectangular rosette Student Strain Gages manufactured by Micro-Measurements Division of Measurements Group, Incorporated. The linear gages employed were EA-06-060LZ-120 linear Student Strain Gages, by the same manufacturer. All strain gages were applied with M-Bond 200 adhesive, also manufactured by Micro-Measurements Division. Panels having a repair applied had three rosettes placed on the front of the panel and a linear gage placed on the rear of the panel (for this investigation the side of the panel facing the front of the test machine will be referred to as the front of the panel, and the reverse side referred to as the rear of the panel). One rosette was placed at the center of the repair area, another above the patch overlap area, and another immediately adjacent to the patch on the original panel surface, as shown in Figure 7. The linear gage on the rear of the panel was

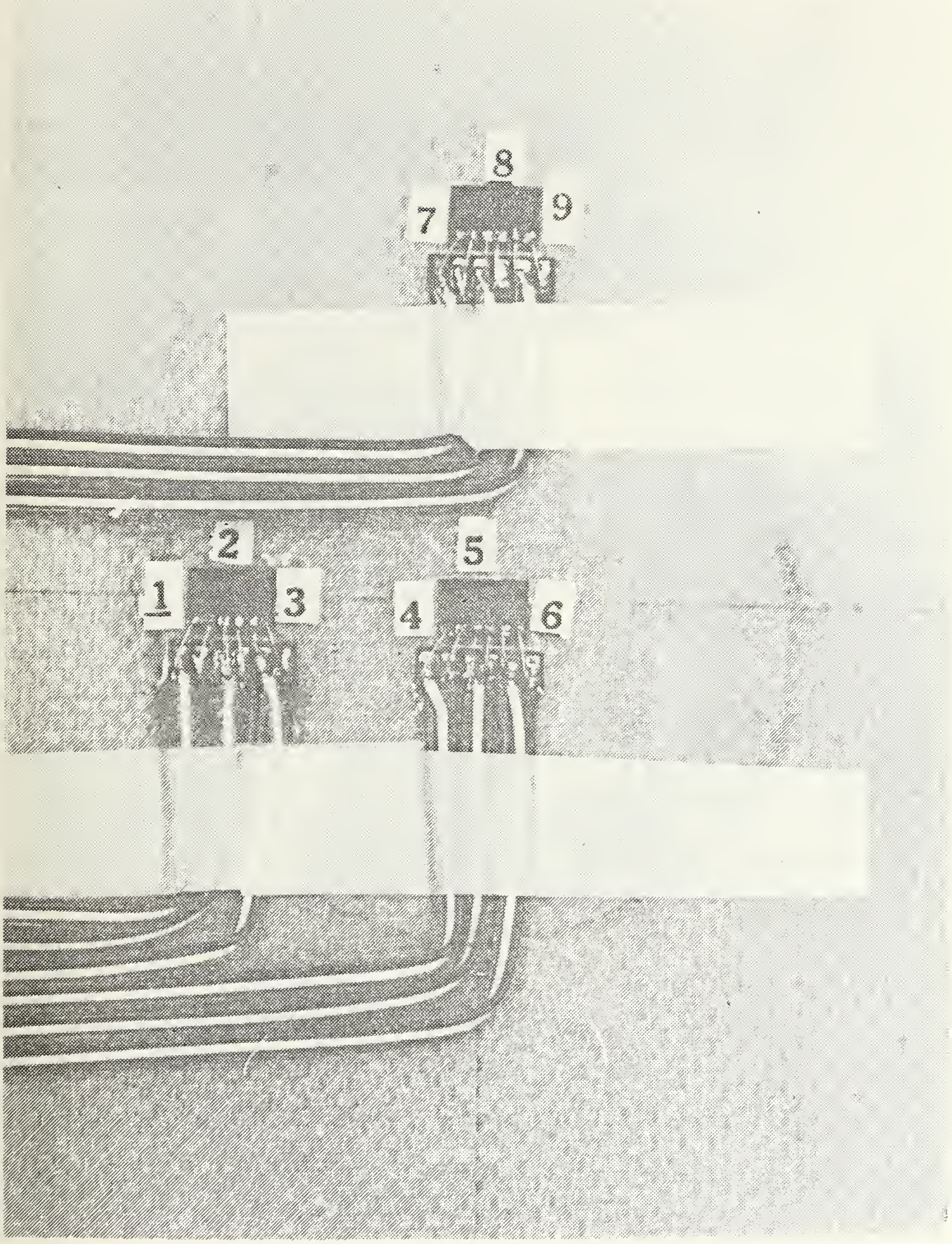


Fig. 7. Location of Rosettes on Sample Patched Panel

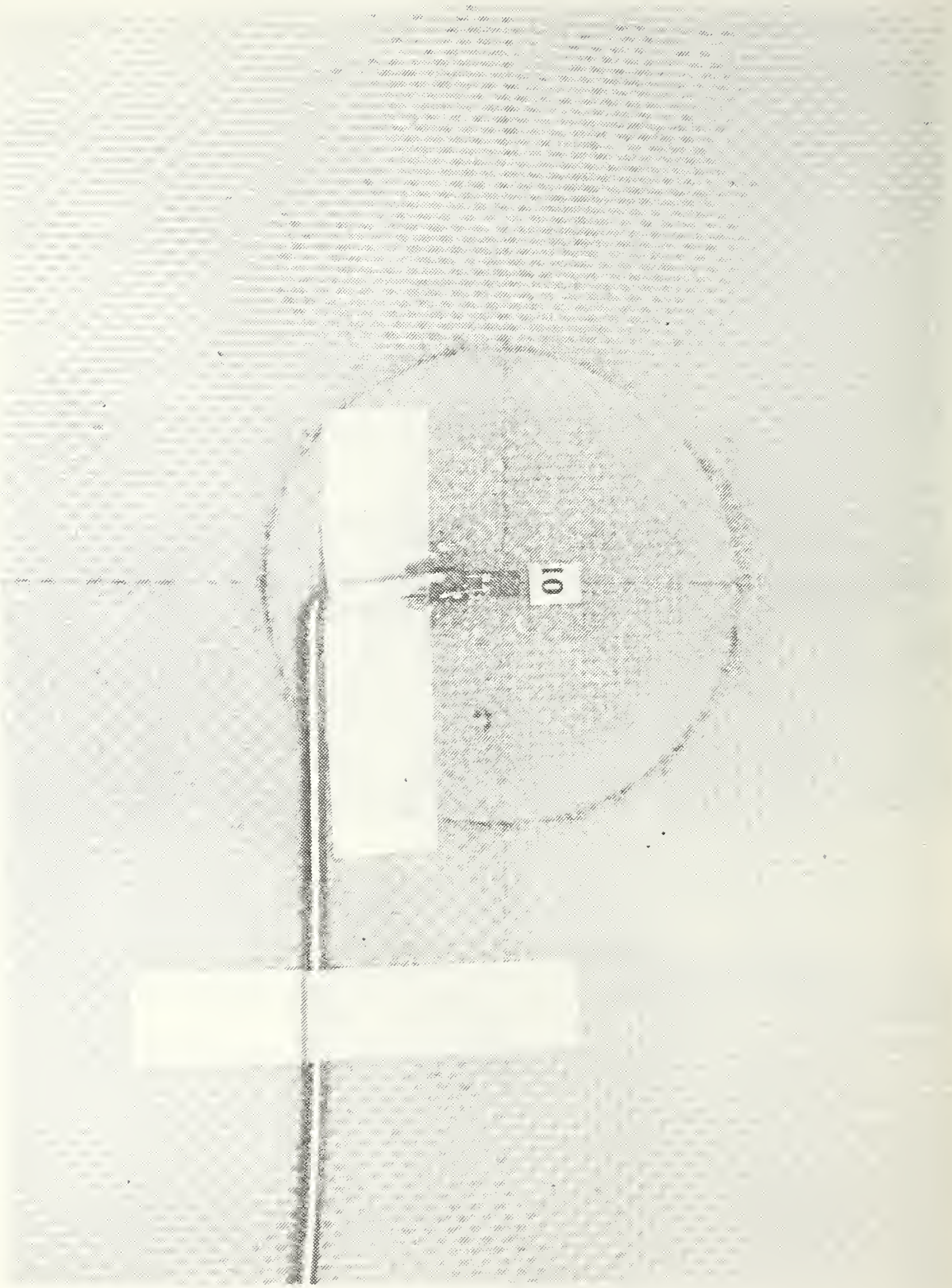


Fig. 8. Location of Linear Gage on Sample Patched Panel

oriented in the horizontal direction, as shown in Figure 8, and was used to indicate the onset of buckling during the loading process. For panels which had no holes, two rosettes were placed on the front of the panel. One was located at the center of the panel, and the other was located 1.50 inches from center along a line drawn horizontally through the center of the panel (in the maximum compression direction). Additionally, a rosette was centered on the rear of the panel, and a linear gage was positioned directly behind the second rosette mounted on the front. For panels which had holes but no repairs, two rosettes were placed on the front of the panel. They were located directly adjacent to the edge of the hole, with one falling on a line drawn vertically through the center of the hole (in the maximum tension direction) and the other falling on a line drawn horizontally through the center of the hole (in the maximum compression direction). A rosette was placed on the rear of the panel immediately behind the vertically oriented rosette on the front, and a linear gage was positioned directly behind the horizontally oriented rosette on the front.

The instrumented panels were secured in the shear fixture, and mounted by means of clevis devices in a 300,000 pound capacity Riehle Testing Machine, as shown in Figure 9. The strain gages were connected to a Micro-Measurements System 4000 Strain Gage Scanner, which was linked to a Hewlett Packard 9825B desktop computer, shown in Figure 10.

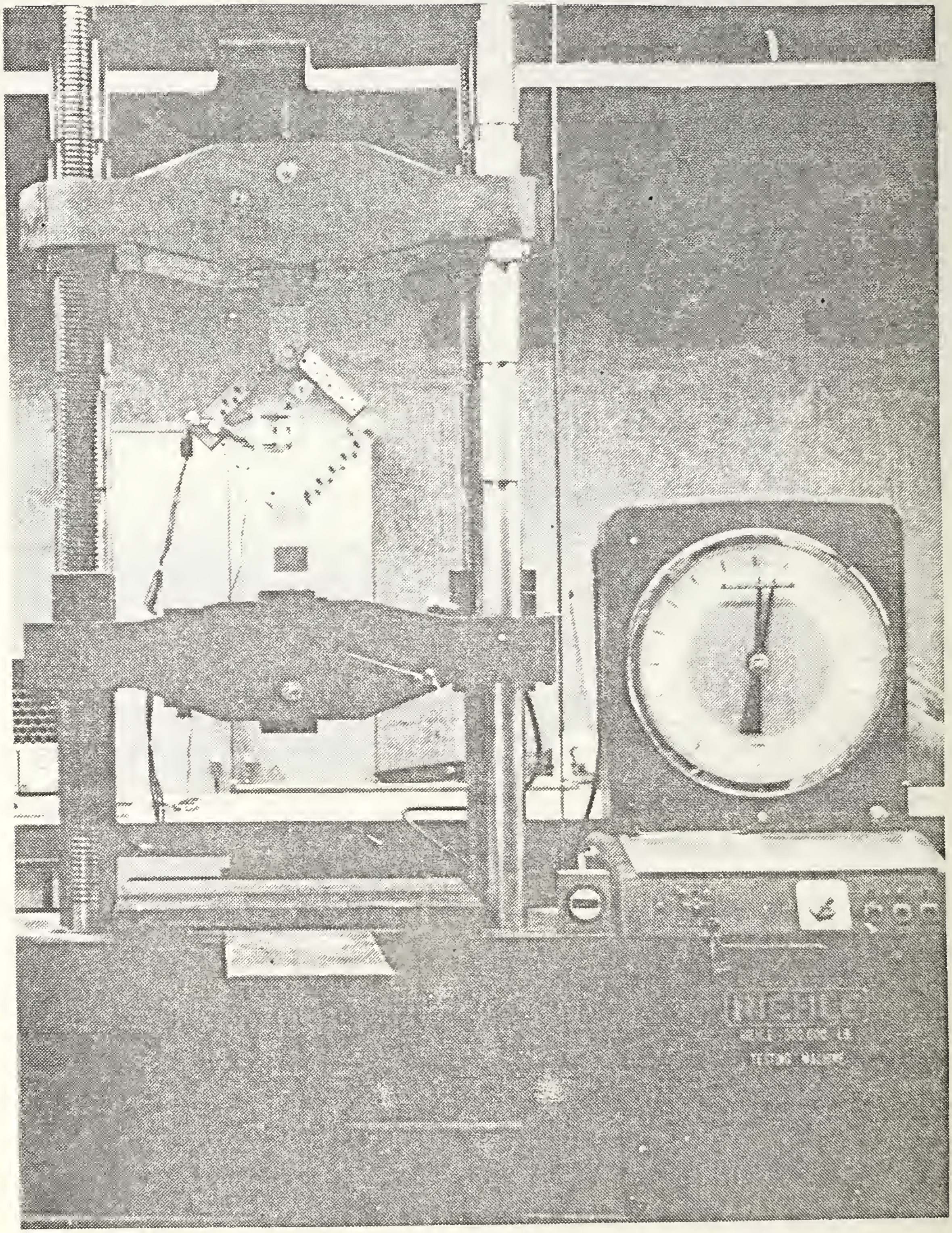


Fig. 9. Riehle Testing Machine

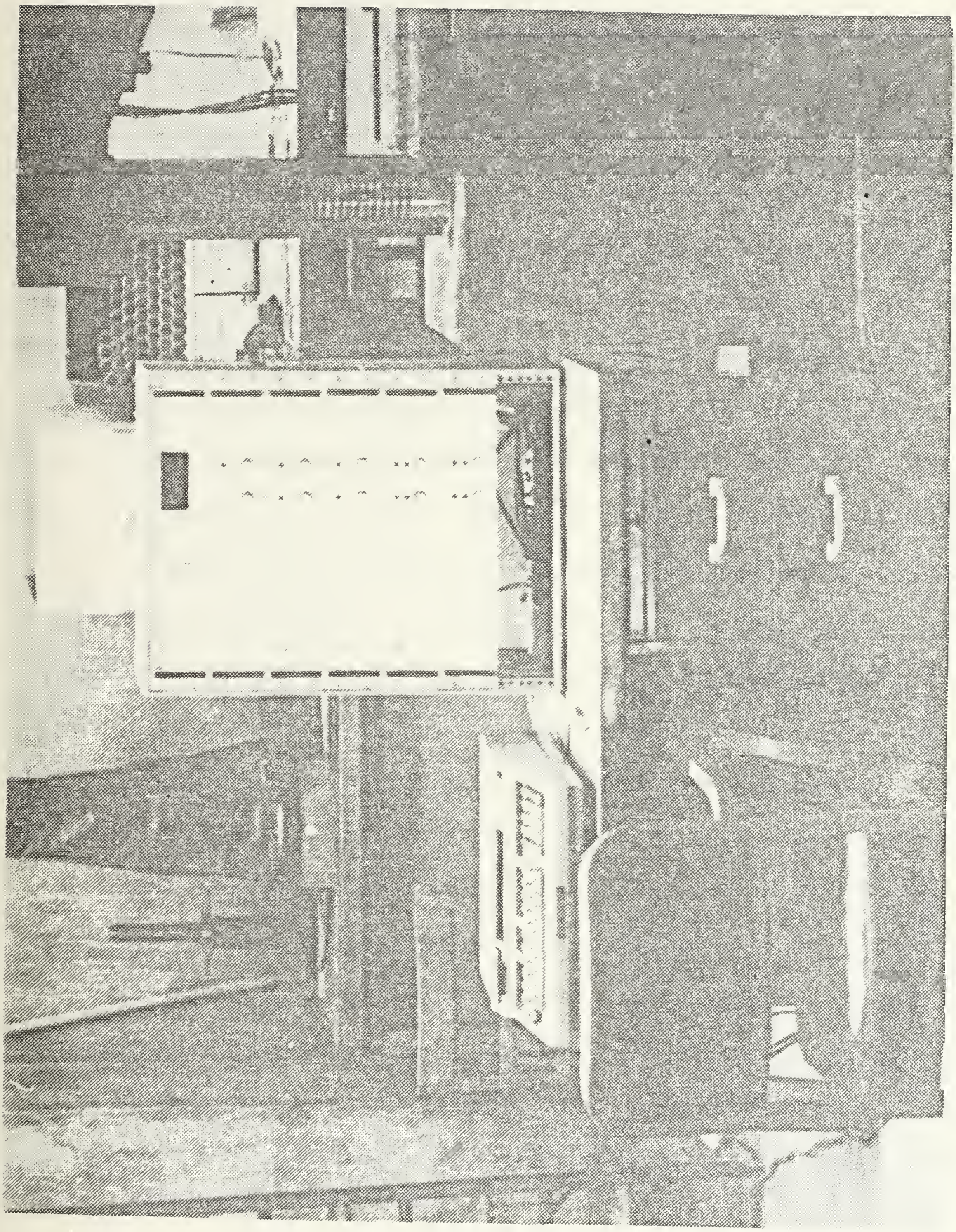


Fig. 10. System 4000 and Hewlett Packard 9825B

D. TEST PROCEDURES

Prior to loading to failure, each panel which had no hole or had a repair applied was loaded to 500 pounds for various system ckecks. Panels with unrepaired holes were loaded to 250 pounds. Loads were then removed and the lower clevis was loosened from the jaws of the testing machine. All strain gages were then zeroed and calibrated, and the zero and calibration values were recorded. The lower clevis was then replaced in the jaws of the testing machine, and a tensile force was applied to the test fixture by the testing machine. The heads of the testing machine were moved apart at a constant rate of 0.025 inches per minute. At 125 pound increments, the time was recorded, as well as the output from each of the strain gages. The System 4000 scans at a rate of approximately 30 channels per second. As only eleven channels were being utilized (three per rosette, one for the linear gage, and one for a temperature compensation gage), the testing machine heads were not stopped for each reading, but were continuously displaced. As the order of magnitude of the smallest time interval between successive 125 pound increments was about 10 seconds, the time taken to scan through the eleven channels resulted in no more than a three percent variance (worst case) in load applied from channel 0 to channel 10 during the scan. For all panels, complete data was taken to the point of failure. For panels

T through Z, load and time were recorded at 125 pound increments between the point of initial failure of the panel and the point of total loss of load carrying capacity.

V. EXPERIMENTAL RESULTS

A. GENERAL RESULTS

The loading and failure process for each of the twenty-six panels followed basically the same sequence. Within a band of a few hundred pounds of 4000 pounds applied load, crackling noises began. These noises continued occasionally until immediately prior to failure of the panel. In general, about two hundred pounds prior to failure, visible deformation of the panel could be seen, similar to that shown in Figure 11. Immediately before failure, a sudden increase of the crackling sound intensity occurred, followed by a loud sharp report at the time of failure. Failure in each case exhibited itself as a vertical buckle running either through or adjacent to the patched area between the opposite tension corners, as depicted in Figure 12.

After failure, panels were removed from the test fixture and inspected. In each case the buckled region exhibited shear failure of the foam core material and no other damage to the Kevlar/epoxy skins. Plots of maximum shear strain versus applied load were prepared (see Appendix B).

Seven panels were continuously loaded following failure. In each case, the crackling sound persisted after failure until the point of total loss of load carrying capability of the panels. At this point, another loud report was heard. Inspection generally revealed cracks originating at the

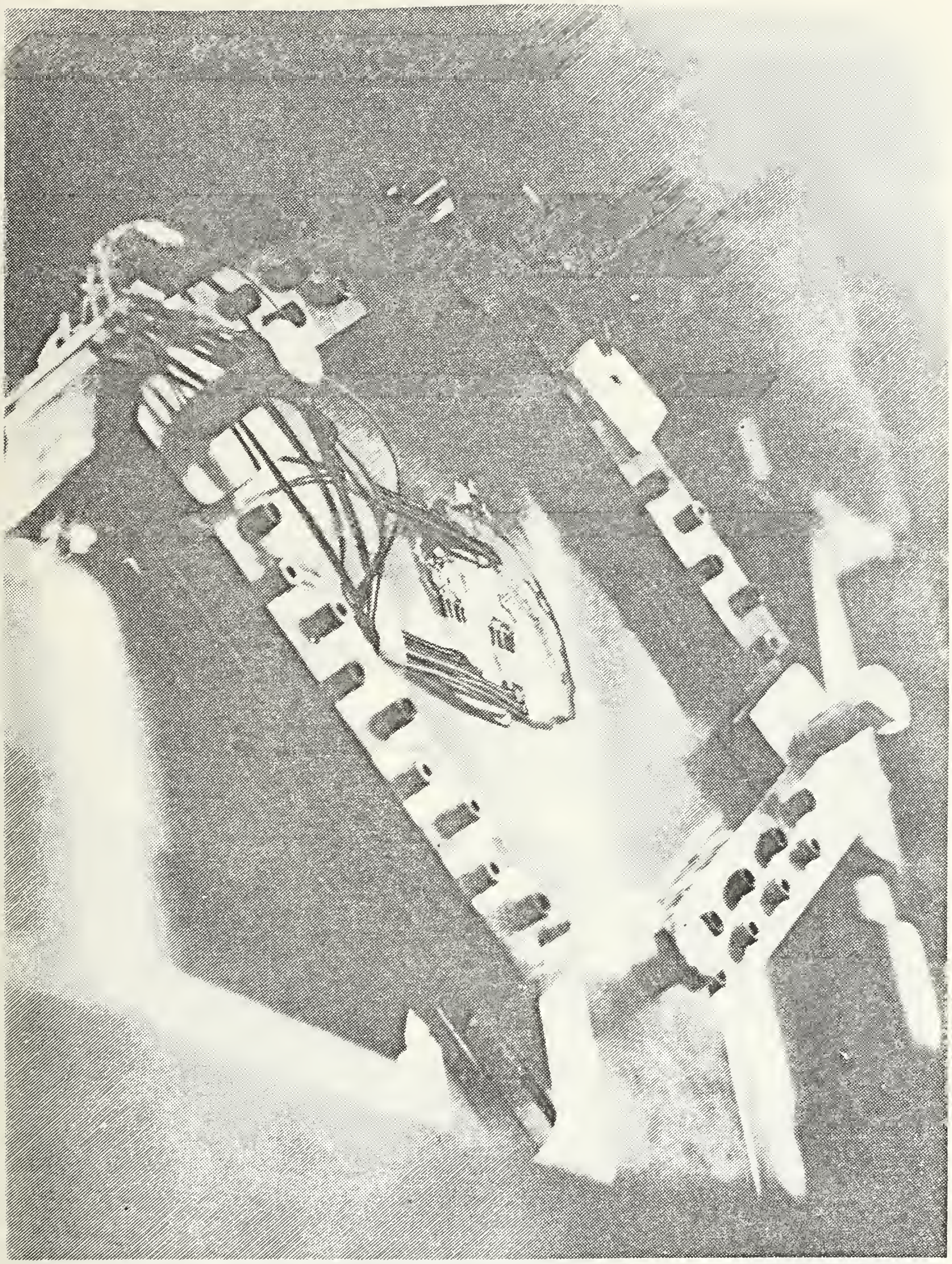


Fig. 11. Deformation of Repaired Panel

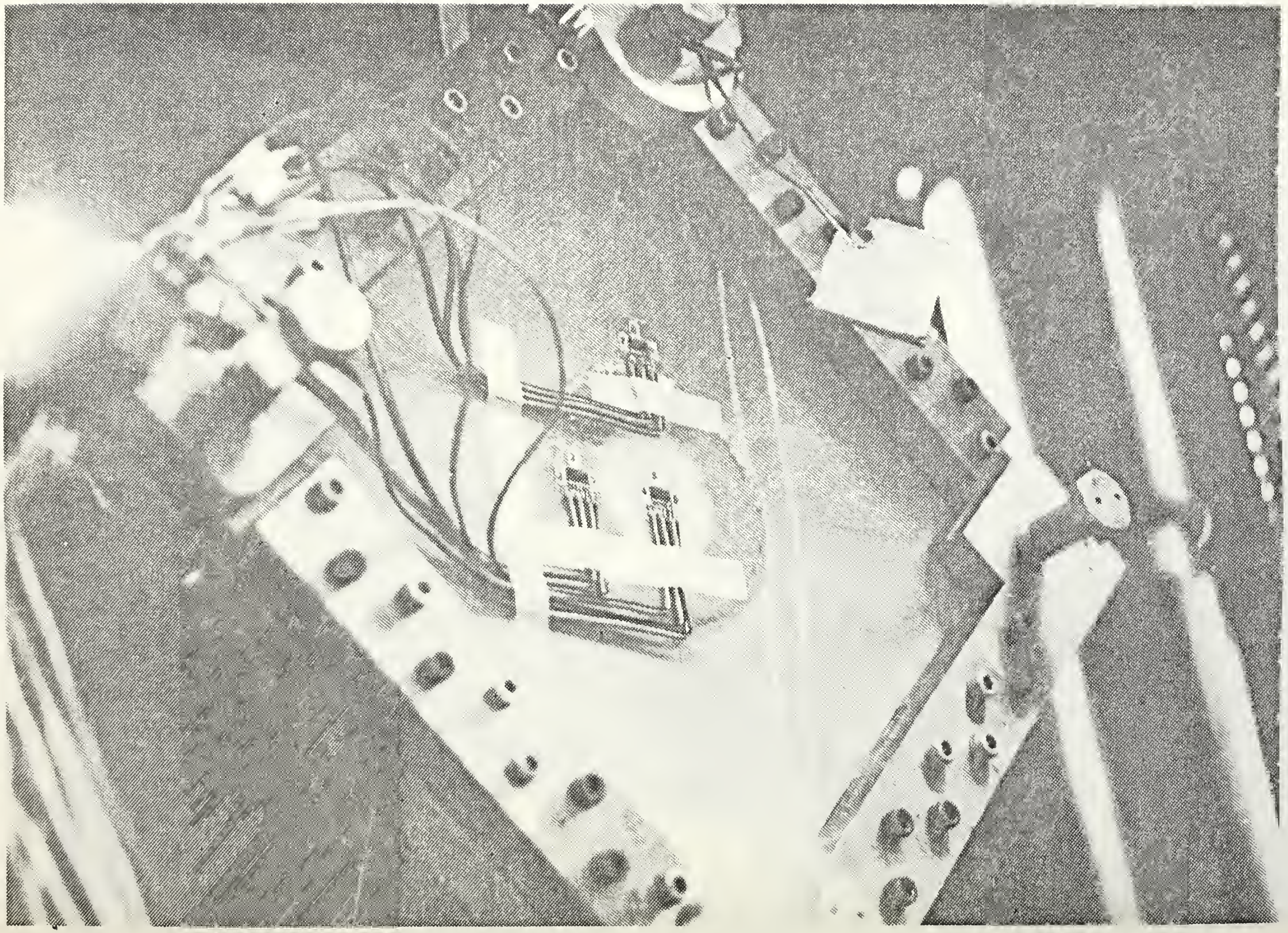


Fig. 12. Vertical Buckle of Repaired Panel

holes in the test area and proceeding horizontally to the compression corners. The cracks penetrated the entire thickness of the panels. Plots of applied load versus total head displacement were prepared. They can be found in Appendix C.

The following twenty-four pages discuss specifically what occurred on each of the individual load cases. A discussion of the results can be found in section VI.

B. BASELINE DATA PANELS

1. Panel A

Beginning at about 5000 pounds applied load, occasional slight crackling sounds were heard. At about 5875 pounds of applied load, a slight bowing of the center of the panel was observed to be oriented vertically. Continued loading resulted in an accentuation of the vertical bowing. At 6300 pounds applied load, a loud report was heard, and the load carried by the panel dropped off sharply. A vertical buckle slightly off center to the right was observed, as shown in Figure 13. In the immediate vicinity of the buckle, the panel was found to have delaminated from the inner core material. The panel was removed from the frame and inspected. A profile view of the buckled region indicated failure of the foam core in shear normal to the plane of the panel. The foam was crushed and pulvarized along the vertical lines of maximum bending of the Kevlar/epoxy layers.

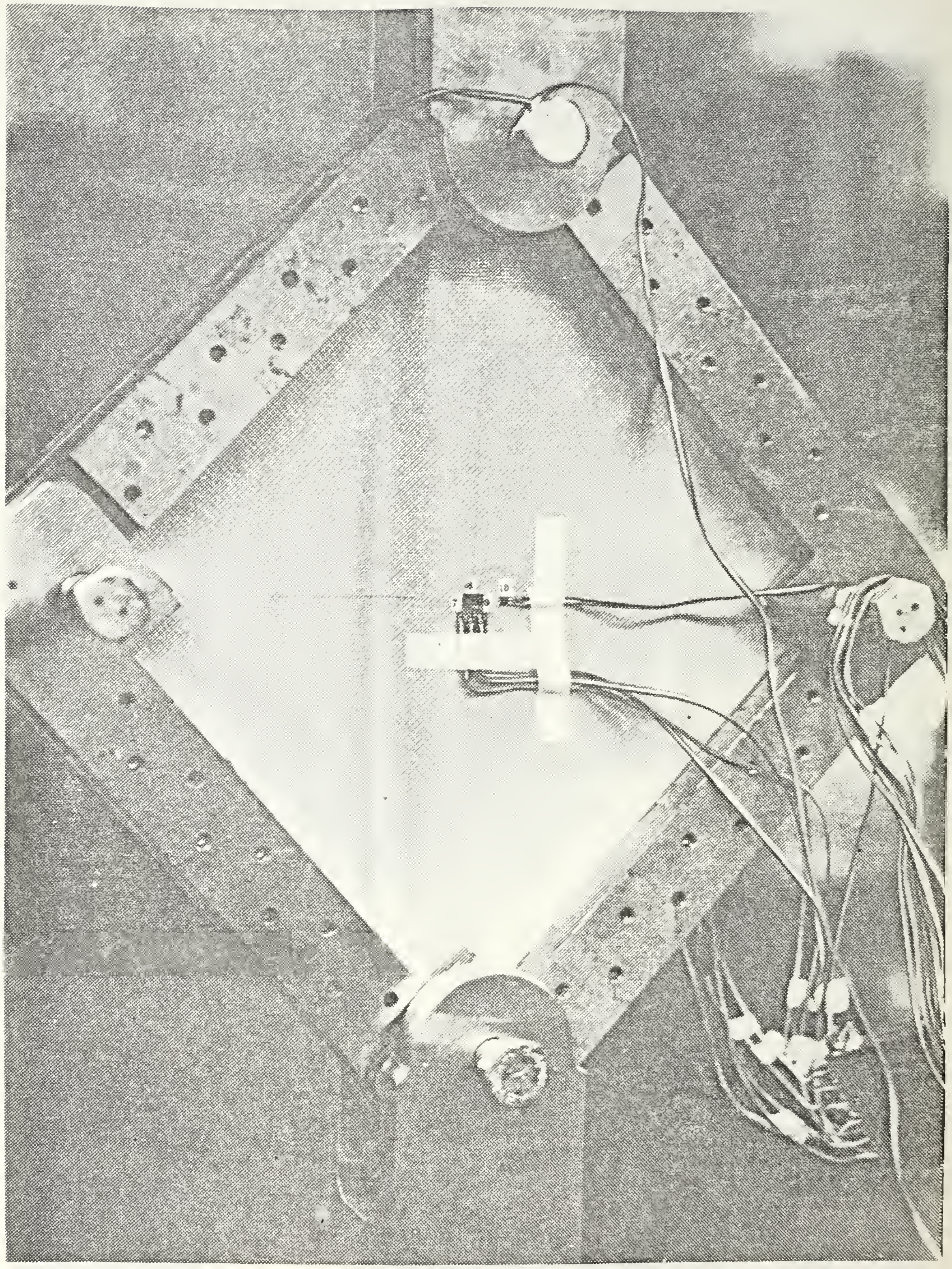


Fig. 13. Vertical Buckle of Panel A, Rear View

Aside from the vertical buckle and its associated delamination, no other damage to the panel was detected.

The maximum shear strain was plotted against the applied load, resulting in a linear curve with shallow slope This graph is presented in Appendix B. From this graph, buckling was determined by noting the radical change in slope. The buckling load was found to be approximately 5900 pounds, with a corresponding maximum shear strain of about 0.005450 in/in. The average shear flow was 464 lbs/in, which exceeded both the limit and the ultimate shear flow requirements.

2. Panel Y

This panel behaved in much the same manner as did Panel A. Slight crackling sounds were heard at about 4500 pounds applied load, and then occasionally throughout the remainder of the loading process. At approximately 5500 pounds applied load, visible deformation was detectable. Vertical bending rapidly developed, culminating in a loud report at 5640 pounds applied load. At this time a distinct, permanent vertical buckle through the center of the panel was observed. At the time of the failure, the load dropped off to slightly less than 4125 pounds. Loading of the panel was not stopped at failure, but continued, during which time load and elapsed times were recorded. Following the initial failure, moderate crackling sounds were audible continuously. The applied load increased at a fairly uniform rate until approximately 6990 pounds, at which time another loud report was heard, and the load dropped to below 1000 pounds. The panel was removed from the test fixture and inspected. The vertical buckle was of the same nature as that in Panel A. Cracks were found connecting the through bolt holes in the tension corners of the panel. These cracks penetrated entirely through the panel. A picture of these cracks appears as Figure 14.

Shear strain was plotted against applied load (see Appendix B), and the buckling load was determined to be 5500 pounds, with a corresponding maximum shear strain of about 0.007890 in/in. The average shear flow was 432 lbs/in. This value exceeded both the limit and the ultimate shear flow requirements. The slope of the curve was slightly greater than that for Panel A, and buckling occurred at about a seven percent lower load.

The heads of the Riehle Testing Machine moved apart at a constant rate of 0.025 inches per minute. By determining the elapsed time between load increments, a plot of load versus total head displacement was obtained. This graph is presented in Appendix C.

3. <u>Panel Z</u>

Similar to Panels A and Y, Panel Z exhibited light crackling sounds beginning at approximately 4875 pounds applied load and continuing periodically until failure. Visible deformation was noted at 5625 pounds, with a loud

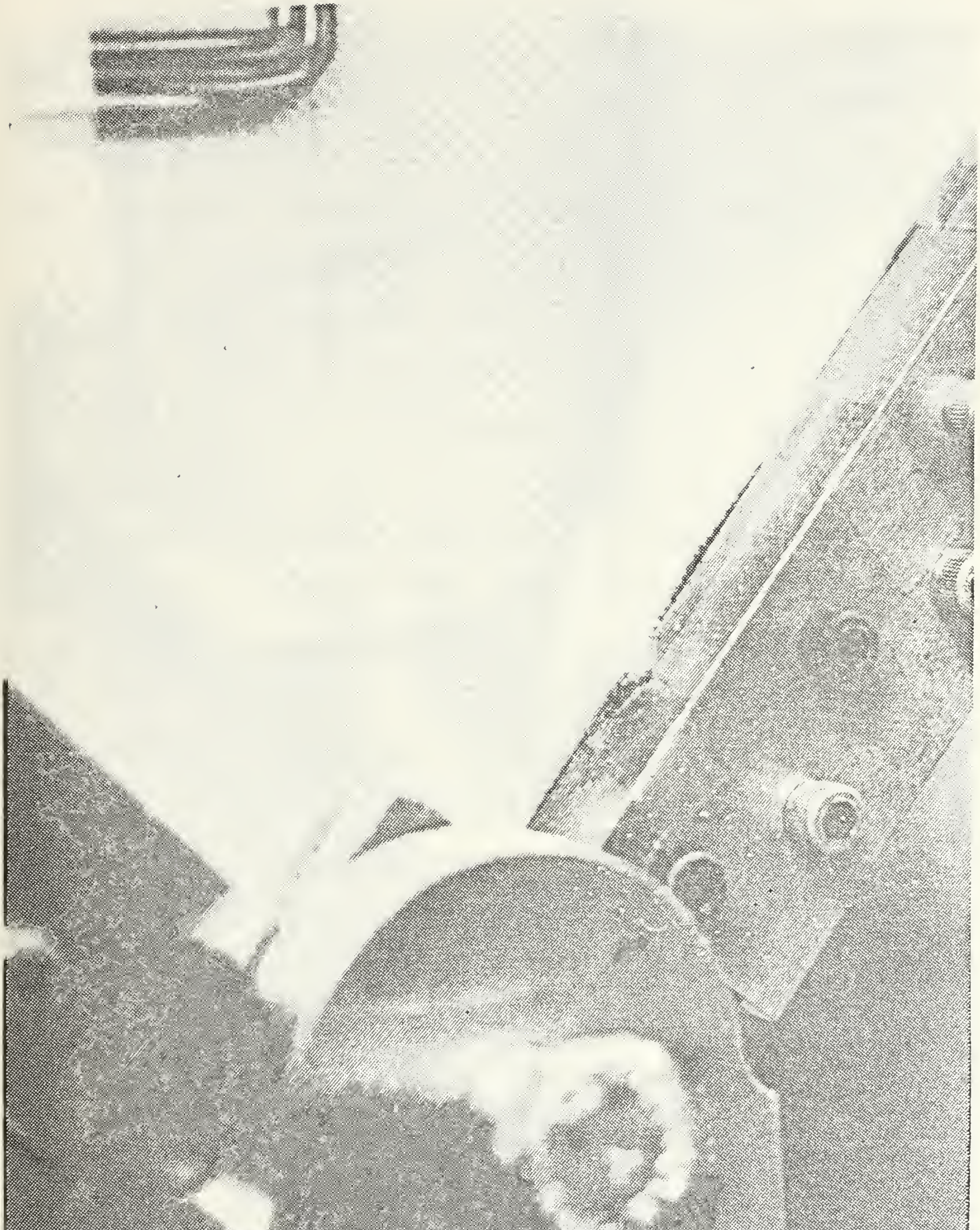


Fig. 14. Crack in Tension Corner of Panel Y

report accompanying large scale buckling at 5850 pounds. A vertical buckle was noted slightly off center to the left of the panel. The load fell off to 4125 pounds at failure. Continued loading was accompanied by constant moderate crackling sounds. During continued loading, a second loud report was heard at 7290 pounds of applied load. At this time the load dropped off to less than 1000 pounds, and a large crack was visible, extending horizontally across nearly half the panel about two inches above the lower tension corner. The panel was removed and inspected. The initial buckle was again characterized by delamination and pulvarizing of the foam core. The horizontal crack was found to have penetrated the entire panel thickness. Figure 15 portrays this crack.

A plot of shear strain versus applied load was prepared (see Appendix B), indicating buckling onset at 5625 pounds, with a corresponding shear strain of approximately 0.006000 in/in. The average shear flow was 442 lbs/in. Again the limit and ultimate shear flow requirements were exceeded. A plot of load versus displacement was prepared and is included in Appendix C.

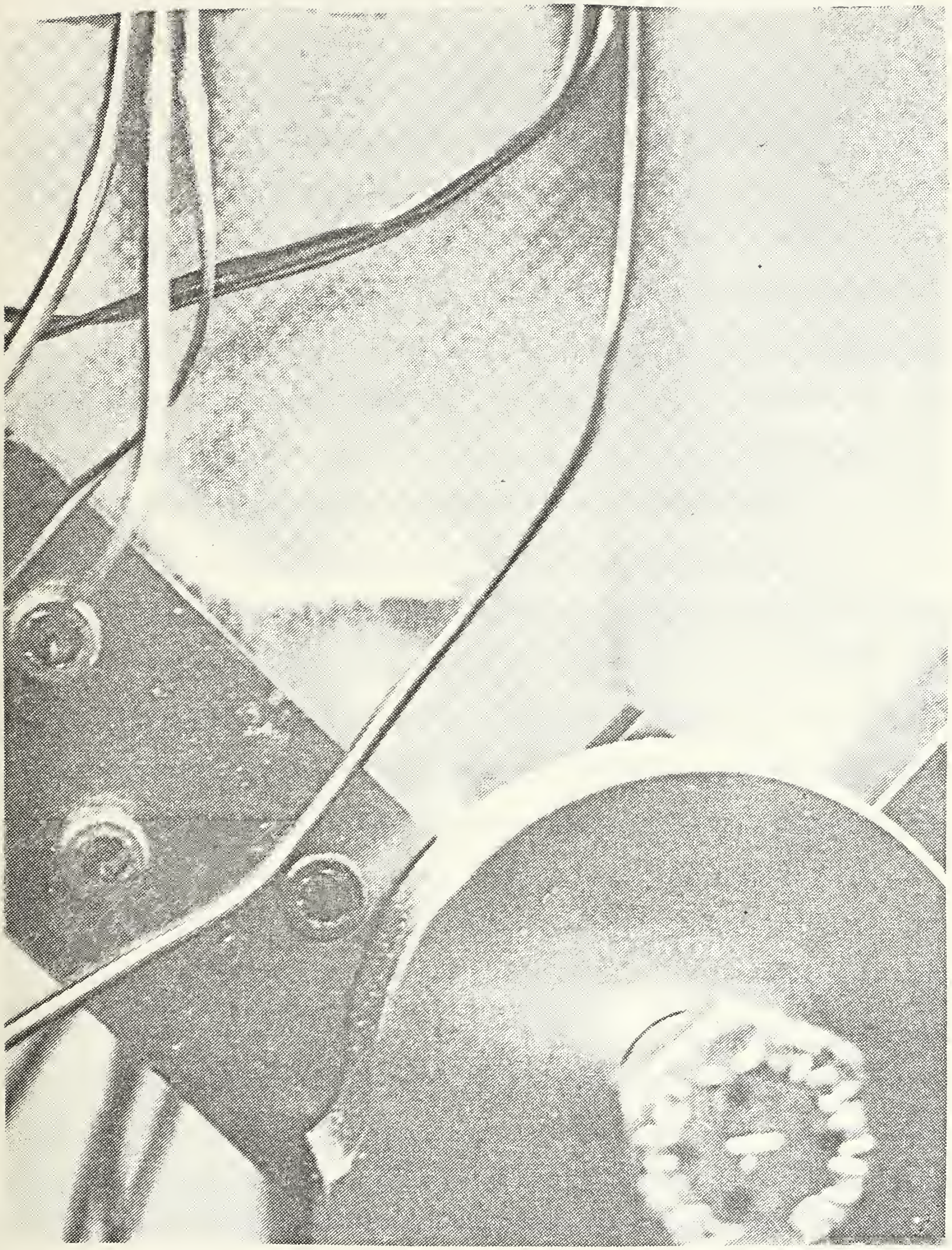


Fig. 15. Crack in Tension Corner of Panel Z

C. 3-INCH PATCHED HOLES

1. Panel B

Slight crackling sounds were detected beginning at about 4000 pounds applied load. Visible deformation was noted at about 5125 pounds, similar to that shown in Figure 11. A loud report occurred at 5425 pounds, accompanying a vertical buckle that was offset to the right of center and just beyond the edge of the patch overlap area, in the same manner as depicted in Figure 12. There was no apparent damage to the patch area itself. After removal from the test fixture, inspection detected shear failure of the core material adjacent to the patch area. The shear strain versus applied load curve (see Appendix B) indicated buckling onset at approximately 5000 pounds, with an accompanying shear strain of 0.015600 in/in. The average shear flow was 393 lbs/in. Again, limit and ultimate shear flow requirements were met.

2. Panel C

Slight crackling sounds began at about 4375 pounds applied load. Visible deformation was detected at 5250 pounds. A loud report occurred at 5575 pounds. At that time a vertical buckle was noted off center to the right and adjacent to the edge of the hole, within the patch overlap area. Post-removal inspection revealed shear failure of the foam core. No other damage was noted.

The shear strain versus applied load curve (see Appendix B) indicated buckling onset at an applied load of 5000 pounds, with a shear strain of 0.012300 in/in. The average shear flow was 393 lbs/in. This exceeded the limit and ultimate shear flow requirements.

3. Panel D

Crackling sounds became audible at about 4125 pounds applied load, with visible deformation being noted at 5625 pounds. A loud report was heard at 5760 pounds. At that time a distinct vertical buckle was noted off center and to the left, adjacent to the edge of the hole and within the patch overlap area. Inspection after removal from the frame revealed shear failure of the foam core and no other damage.

Buckling onset was determined to occur at 5500 pounds applied load by the shear strain versus applied load curve (see Appendix B). The shear strain at that load was about 0.016970 in/in. The average shear flow was 432 lbs/in. Again, limit and ultimate shear flow requirements were exceeded.

D. 2-INCH PATCHED HOLES

1. Kevlar Patches with Foam Plugs

a. Panel E

Slight crackling sounds emanated from the panel, beginning at about 4250 pounds of applied load, and continuing periodically until failure, which occurred at a load of 5780 pounds. A loud report accompanied the buckling failure. Deformation had been visually detected at about 5500 pounds of applied load. The resultant buckle extended vertically from the upper to the lower corner, and was displaced to the left, just outboard of the patch overlap area. Inspection after removal from the "picture frame" revealed shear failure of the foam core material.

The shear strain versus applied load curve (see Appendix B) indicated the onset of buckling to have occurred at 5375 pounds of applied load, with a maximum shear strain at that point of approximately 0.012730 in/in. The average shear flow was 422 lbs/in. Limit and ultimate shear flow requirements were exceeded.

b. Panel F

Upon loading, slight crackling noises were detected, beginning about 4125 pounds of applied load. At 4875 pounds, a vertical wrinkle became visible in the patch itself on the front side of the panel. The wrinkle was about 0.75 inches off center to the left, and extended over the foam plug, but did not extend over the patch overlap area. A small amount of delamination was noted in the immediate vicinity of the wrinkle. The panel itself exhibited no visible deformation until 5250 pounds. A loud report was heard at 5500 pounds, at which time a vertical buckle formed. It was located slightly off center to the left, just inboard of the edge of the foam plug. Post-removal inspection revealed shear failure of the foam core material.

The plot of shear strain versus applied load (see Appendix B) indicated the onset of buckling at 5000 pounds applied load, with about 0.005975 in/in maximum shear strain. Average shear flow was 393 lbs/in. Limit and ultimate shear flow requirements were exceeded.

c. Panel G

Initial crackling noises were heard at a load of 4375 pounds. Visible deformation was noted at 5750 pounds. At 6170 pounds, a loud report sounded, and a vertical buckle was noted off center to the left. It was located outboard of the foam plug area, about halfway through the overlap area. The panel was removed from the test fixture and inspected. Shear failure of the foam core was found to have occurred.

The shear strain versus applied load curve (see Appendix B) showed the onset of buckling to have occurred at about 5125 pounds applied load, with an accompanying maximum shear strain of 0.011890 in/in. The average shear flow was 403 lbs/in, exceeding both limit and ultimate shear flow requirements.

2. Fiberglass Patches with Foam Plugs

a. Panel H

Slight crackling noises were detected at 4000 pounds of applied load, and continued intermittently through failure of the panel. At about 5750 pounds of applied load, crazing of structural adhesive at the center of the rear patch became visible. No visible deformation was noted prior to failure, which occurred at 6375 pounds of applied load. A sharp loud report accompanied the failure, which exhibited itself as a vertical buckle lying just outboard of and adjacent to the edge of the patch overlap. Inspection of the panel after removal from the "picture frame" revealed shear failure of the foam core material.

The plot of maximum shear strain versus applied load (see Appendix B) indicated the onset of buckling at 5750 pounds, with an accompanying maximum shear strain of approximately 0.010180 in/in. Average shear flow was 452 lbs/in. Both limit and ultimate shear flow requirements were exceeded.

b. Panel I

During loading, slight crackling sounds were audible from about 4375 pounds applied load to failure at intermittent intervals. Crazing became visible on the front and rear patches at 5500 pounds. No visible deformation was noted prior to failure. A loud report was heard at an applied load of 6625 pounds, and a vertical buckle appeared, located just outboard of and adjacent to the left edge of the patch overlap area. After removal, inspection of the panel revealed shear failure of the foam core.

Maximum shear strain was plotted against applied load (see Appendix B), and no significant change in slope was noted prior to failure. This indicated that the onset of buckling occurred at the failure load of 6625 pounds, with a maximum shear strain of about 0.010380 in/in. Average shear flow was 521 lbs/in. Both limit and ultimate shear strain requirements were exceeded.

c. Panel J

Crackling noises initiated at about 4375 pounds of applied load. Crazing was noted on the rear patch at 5375 pounds and on the front patch at 5625 pounds. Deformation of the panel became visible at 6000 pounds. A loud report sounded at 6270 pounds, accompanied by a vertical buckle which ran adjacent to the outer edge of the patch overlap area right of center. Post-removal inspection revealed shear failure of the foam core material.

The onset of buckling was determined from the plot of maximum shear strain versus applied load (see Appendix B) to have occurred at 5500 pounds, with 0.010435 in/in maximum shear strain. The average shear flow was 432 lbs/in. Limit and ultimate shear flow requirements were both exceeded.

3. Kevlar Patches with Structural Adhesive Plugs

a. Panel K

At an applied load of 2250 pounds, the front patch began to delaminate from the plug area. The overlap area remained intact. At 3000 pounds, a vertical wrinkle formed in the center of the patch. Crackling noises were noted intermittently from about 4250 pounds until failure. Deformation of the panel became visible at 5125 pounds applied load. At 5590 pounds, a loud sharp report was heard, and a distinct vertical buckle was noted to the left of center, adjacent to the patch overlap area. Inspection after removal from the "picture frame" revealed shear failure of the foam core material. Aside from that and the delamination of the patch on the front over the plug, no other damage was noted.

Plotting maximum shear strain against applied load (see Appendix B) resulted in a buckling onset load of about 4625 pounds, with 0.006630 in/in maximum shear strain. The average shear flow was 363 lbs/in. This exceeded both the limit and the ultimate shear flow requirements.

b. Panel L

Crackling sounds became audible at an applied load of 4125 pounds, and continued sporadically until panel failure. Visible deformation of the panel was noted at 5500 pounds. A loud report accompanied by a vertical buckle indicated failure at 6190 pounds. The buckle ran vertically, adjacent to the left edge of the patch overlap area. After removal, inspection revealed shear failure of the foam core material.

The onset of buckling was determined to have occurred at an applied load of 5325 pounds from the maximum shear strain versus applied load curve (see Appendix B). The shear strain at this point was approximately 0.008330 in/in. The average shear flow was 418 lbs/in, exceeding both the limit and ultimate shear flow requirements.

c. Panel M

A delamination of the patch from the adhesive plug on the front side of the panel began at an applied load of 2250 pounds. At 3750 pounds, a vertical wrinkle became visible in the patch above the plug. No damage or distortion was noted on the rear patch. At 4125 pounds, slight crackling noises were heard, which continued occasionally until failure. Visible deformation was noted at 5375 pounds, and at 5900 pounds a loud report was heard. At this time a vertical buckle was noted left of center, adjacent to the patch overlap area. Post removal inspection revealed shear failure of the foam core material.

The plot of maximum shear strain versus applied load (see Appendix B) indicated buckling onset at 5500 pounds, with approximately 0.008230 in/in maximum shear strain present. Average shear flow was 432 lbs/in. This met both the limit and the ultimate shear flow requirements.

4. <u>Kevlar Patches with no Plugs</u>

a. Panel N

At an applied load of nearly 1000 pounds, the front patch began to visibly deform over the area of the hole. By approximately 1625 pounds, a clearly discernible vertical wrinkle extended through the center of the front patch. The patch overlap area, as well as the entire rear patch, remained undeformed. At about 4000 pounds, a rapid onset of moderate crackling sounds was heard, followed at 4070 pounds by a loud report. At that time a vertical buckle was noted extending from the upper tension corner through the center of the panel to the lower tension corner. Upon removal from the "picture frame" it was inspected, revealing shear failure of the core material.

The maximum shear strain versus applied load curve (see Appendix B) indicated a change in slope at 1000 pounds, corresponding to the initiation of the wrinkle on the front patch. Buckling was determined to have occurred at approximately 3875 pounds, at which point the maximum shear stress was about 0.008850 in/in. Average shear flow was 304 lbs/in. The requirement for limit shear flow was exceeded.

b. Panel O

At an applied load of 1875 pounds, a loud report was heard. A delamination of the rear patch had occurred from the five o'clock position to the eight o'clock position. The load did not drop off appreciably. At 3000 pounds, a second loud report was heard, and the rear patch delamination had spread to the region from the four o'clock position to the ten o'clock position, fully half the circumference of the patch. Again the load dropped off only slightly. At nearly 3500 pounds, a rapid onset of crackling occurred, followed immediately by a third loud report. At this time the load dropped off significantly. A vertical buckle was noted through the center of the panel. Inspection following removal from the shear fixture revealed shear failure of the foam core material.

The plot of shear strain versus applied load (see Appendix B) indicated a radical change in slope at an applied load of 2000 pounds, corresponding to the delamination of the rear patch. Any data beyond that point was not relevant to this investigation. At the time of the rear patch delamination, the maximum shear strain was found to be approximately 0.004550 in/in. Average shear flow was 157 lbs/in, which exceeded limit shear flow requirement.

c. Panel P

At an applied load of 3875 pounds a slight crackling noise became audible. The sound persisted intermittently through about 4250 pounds, at which time it intensified and became continuous. At 4325 pounds, a sharp loud report was heard, and a vertical buckle was noted through the center of the panel. Until failure, no deformation had been visible. Inspection of the panel after it had been removed from the test fixture revealed shear failure of the foam core material.

The plot of maximum shear strain versus applied load (see Appendix B) showed no significant change in slope prior to failure. The buckling load was the failure load of 4325 pounds, with a corresponding maximum shear strain of approximately 0.010775 in/in. The average shear flow was 340 lbs/in. The limit shear flow requirement was exceeded.

E. 1/2-INCH PATCHED HOLES

1. Panel Q

Slight crackling sounds became audible at an applied load of about 4125 pounds, and continued periodically through the loading cycle to failure. At 5500 pounds, visible deformation of the panel was noted. At 5820 pounds, a loud report was heard, and a vertical buckle was noted slightly off center to the left. Once removed from the shear fixture, inspection of the panel revealed shear failure of the foam core material. No other damage was noted.

The plot of maximum shear strain versus applied load (see Appendix B) showed the onset of buckling at 5500 pounds, with a corresponding shear strain of 0.015680 in/in. The average shear flow was 432 lbs/in. Both limit and ultimate shear flow requirements were exceeded.

2. Panel R

At 4250 pounds of applied load, slight crackling sounds became audible. These sounds continued intermittently until failure occurred. Deformation of the panel became visible at 5750 pounds. At 6075 pounds, a loud report was heard, and a vertical buckle was noted through the patch overlap area left of center. Post-removal inspection of the panel revealed shear failure of the foam core material.

The buckling load determined from the shear strain versus applied load curve (see Appendix B) was about 5750 pounds, with a corresponding maximum shear strain value of approximately 0.009595 in/in. Average shear flow was 450 lbs/in. This exceeded both the limit and the ultimate shear flow requirements.

F. UNPATCHED HOLES

1. Panel S

Slight crackling sounds became audible at an applied load of about 4000 pounds. These sounds continued sporadically until about 5625 pounds, at which time they became continuous and somewhat louder. At that load, visible deflection of the surface of the panel was noted. At 5775 pounds, a loud report was heard and a vertical buckle was noted in the center of the panel. Inspection after removal of the panel from the frame revealed shear failure of the foam core material.

The shear strain versus applied load curve (see Appendix B) displayed no significant change in slope prior to failure, resulting in a buckling load of 5775 pounds. The corresponding maximum shear strain was approximately 0.008900 in/in. Average shear flow was 450 lbs/in, which exceeded both the limit and the ultimate shear flow requirements.

2. Panel T

During the loading process, nothing visible or audible was noted until just beyond 4000 pounds applied load, at which time a continuous crackling sound of moderate intensity suddenly began. At 4045 pounds, a sharp loud report was heard, at which time the load dropped to 3250 pounds. A vertical buckle was noted through the center of the panel. Load was continuously applied following failure, during which time continued crackling sounds were heard. At 4315 pounds, a second report was heard, and the load dropped to less than 500 pounds. Horizontal cracks were observed on front and rear of the panel on both left and right sides of the hole. After removal from the "picture frame", inspection revealed shear failure of the core material along the buckle. The horizontal cracks were found to have penetrated entirely through the thickness of the panel.

A plot of maximum shear strain versus applied load (see Appendix B) indicated the onset of buckling at about 3000 pounds applied load and 0.003830 in/in shear strain. Average shear flow was 236 lbs/in. This exceeded the limit shear flow requirement. A plot of applied load versus total head displacement is shown in Appendix C, with a sharp discontinuity corresponding to the buckling failure load. Average shear flow at the time of the cracking was 339 lbs/in, or 3 lbs/in less than the ultimate shear flow requirement.

3. Panel U

Slight crackling sounds were heard at 4125 pounds applied load, and continued periodically until failure. At 5150 pounds, a sudden increase in the crackling sound was followed immediately by a loud sharp report. A vertical buckle was observed through the center of the panel. Continued loading resulted in continuation of the crackling sound until a second loud report was heard at 5560 pounds. At that time cracks were seen to extend horizontally through the center of the panel, on the front and rear, to the left and right of the hole. The load dropped off to less than 625 pounds. Post-removal inspection revealed shear failure of the foam core material in the vicinity of the vertical buckle. The cracks were found to have penetrated the entire thickness of the panel.

Maximum shear strain was plotted against applied load (see Appendix B), indicating a buckling load of about 4875 pounds. The corresponding maximum shear strain was approximately 0.005450 in/in. Average shear flow was 383 lbs/in, exceeding both limit and ultimate shear flow requirements. A distinct discontinuity in the applied load versus total head displacement curve (see Appendix C) corresponded to the failure load of the panel.

4. Panel V

During the loading process, a sudden onset of moderate crackling sounds was immediately followed by a sharp loud report at an applied load of 3210 pounds. The load dropped off to slightly less than 2250 pounds. A vertical buckle was noted through the center of the panel above and below the hole. The loading process was continued and the crackling sound persisted. At 3445 pounds, another loud report was heard, and the load dropped to less than 500 pounds. Cracks were observed to the right and left of the hole on the front and rear of the panel. Inspection of the panel after it was removed from the test fixture revealed shear failure of the foam core material in the region of the buckle. The horizontal cracks were found to have penetrated the entire thickness of the panel.

A plot of shear strain versus applied load (see Appendix B) showed no significant change in slope. The buckling load was 3210 pounds, with a maximum shear strain at that point of 0.005850 in/in. Average shear flow was 252 lbs/in, exceeding the limit shear flow requirement. Plotting applied load against total head displacement showed a discontinuity at the buckling load (see Appendix C). At the point of loss of load carrying capability, the average shear flow was 265 lbs/in, still well short of the ultimate shear flow requirement.

5. Panel W

A sudden onset of moderately loud crackling sound was immediately followed by a loud report at 2490 pounds applied load. The load dropped to slightly less than 1625 pounds, and a vertical buckle was noted above and below the hole in the center of the panel. The crackling sound continued as the load was increased, and a second loud report was heard at 2950 pounds. The load dropped to less than 375 pounds, and horizontal cracks appeared to the right and left of the hole on both the front and the rear of the panel. The panel was inspected after removal from the "picture frame", revealing shear failure of the foam core material in the vicinity of the vertical buckle. The crack had penetrated the entire thickness of the panel.

No change in slope was discernible on the shear strain versus applied load curve (see Appendix B), and the buckling load was determined to be 2490 pounds, with an accompanying maximum shear strain of approximately 0.007550 in/in. Average shear flow was 196 lbs/in. This value exceeded the requirement for limit shear flow. A discontinuity on the applied load versus total head displacement curve corresponded to the buckling load of the panel (see Appendix C). Average shear flow at the time of loss of load carrying capacity was 232 lbs/in, well short of the ultimate shear flow requirement.

6. Panel X

At 1700 pounds applied load, a sudden loud report was heard. A vertical buckle was noted along the centerline of the panel from the lower edge of the hole to the lower tension corner. The load dropped to about 1500 pounds. At 1550 pounds a second report was heard, and a vertical buckle was observed from the top of the hole along the panel centerline to the upper tension corner. As the load was increased, crackling sounds were heard, and a third loud report was heard at 2560 pounds. At this time horizontal cracks were noted from the edges of the hole to the compression corners of the frame on both the front and the rear of the panel, as shown in Figure 16. Post-removal inspection revealed shear failure of the foam core material in the regions of the vertical buckles. The cracks were found to have penetrated the entire thickness of the panel.

The maximum shear strain versus applied load curve (see Appendix B) displayed no noticeable change in slope. The buckling load was determined to be 1700 pounds, with an accompanying maximum shear strain of 0.005015 in/in. Average shear flow was 134 lbs/in, exceeding the limit shear flow requirement. The applied load versus total head displacement curve (see Appendix C) showed a discontinuity corresponding to the failure load of the panel. At the time of total loss of load carrying capacity, the average shear flow was 201 lbs/in, well short of the ultimate shear flow requirement.

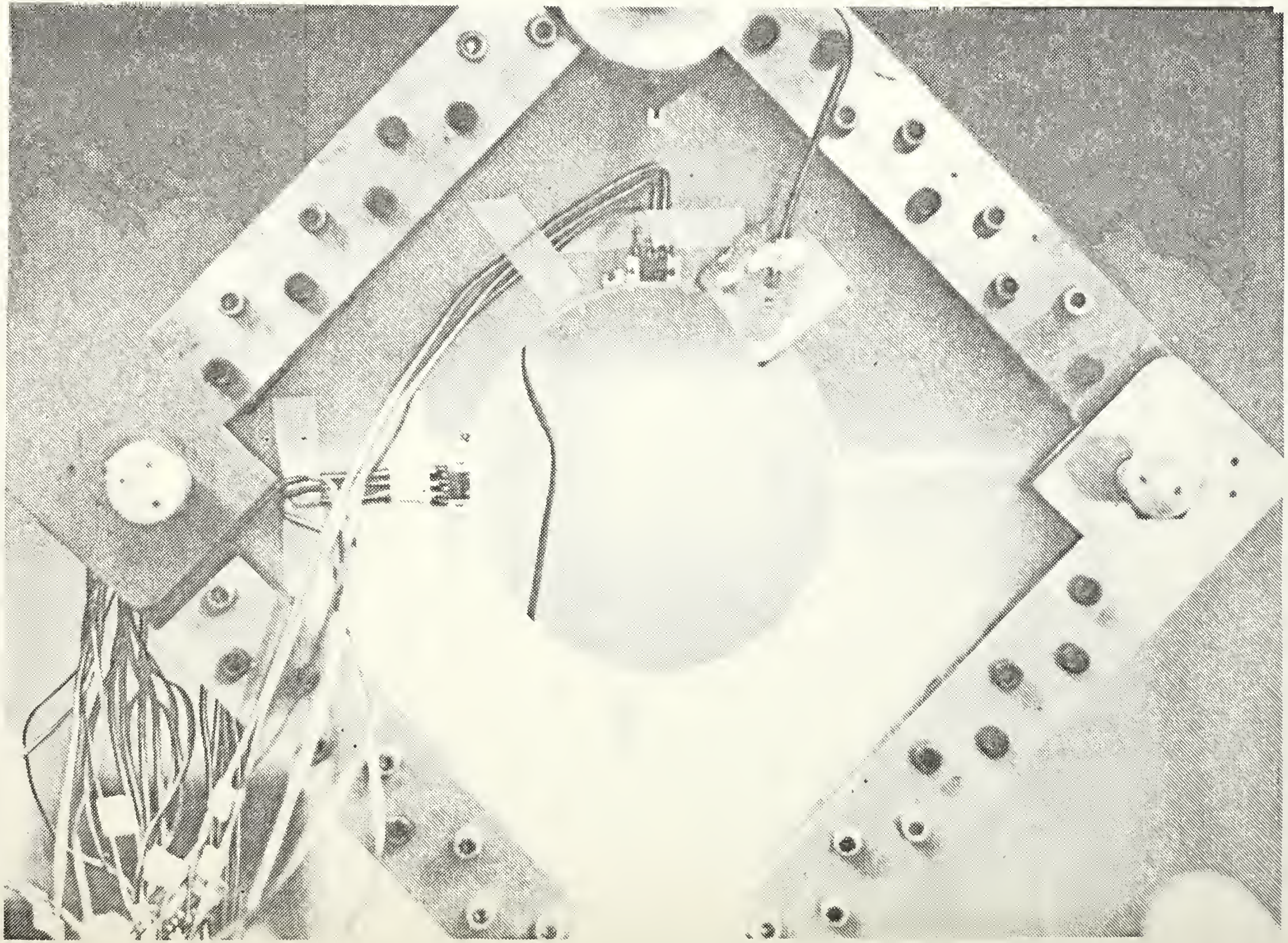


Fig. 16. Vertical Buckle and Horizontal Cracks on Panel X

VI. DISCUSSION OF RESULTS

A. ASSESSMENT OF THE TEST FIXTURE

Almost invariably there was a small non-linear region at very low load levels for the shear strain versus applied load curves located in Appendix B. Without exception the curves became linear within 500 pounds of applied load (only three panels exhibited non-linearity beyond 250 pounds). Close examination of the photoelastic panel photogtaphs in Appendix A show that some non-uniformity in stress distribution does in fact occur at low load levels, but is continually "pushed" into the corner regions of the test area as the applied load increases.

There are several possible reasons for the variance from linearity at low load levels. Initially, when the jaws of the Riehle Testing Machine grip the clevis assemblies, some slippage may occur as the knurled jaw faces bite into the aluminum clevises. The slippage will give the appearance of less strain under the particular loading situation than there would otherwise be.

A second reason for the non-linearity is the acual accuracy of the load measuring system at the low load levels. The Riehle Testing Machine employed has a load capacity of 300,000 pounds, with a single adjustable load measurement system for the entire range of loads. Calibration data for the load indicating system was not available for the range of loads encountered in this investigation.

Slippage of the sample panel within the test frame would also yield results slightly different than ideal. Additionally, and initial stress loads resulting from the method of securing the panel in the "picture frame" would give rise to variations which would become less critical as the general state of stress of the panel was increased with increasing load.

Another possible source of the variation from linearity is the unloaded geometric shape of the panel being tested. If the panel had a slight cutvature prior to being loaded in the test fixture, the initial application of load would tend to flatten the panel. Deformation would then occur at a higher rate than once the panel became planar.

The buckling modes for all twenty-six panels were consistent with one another. In each case a vertical buckle appeared, perpendicular to the direction of maximum compression. The buckles were observed to be confined to the test area of the panels, not extending into the area of the panel clamped into the test fixture. There was no apparent separation or stretching of the panel material in the compression corners, nor was there any scissoring of the panel material in the tension corners. The problems inherent in "picture frame" shear fixtures where panel test area corners do not coincide with the corner pivots of the test fixture seem to have been eliminated by the apparatus design employed in this investigation.

All panels were tested in the same shear fixture with the same clevis assemblies. The shear fixture was assembled in precisely the same manner for each of the twenty-six test specimens. This further reduced the amount of error that could be expected when testing a sample population of this size.

B. ASSESSMENT OF PANEL FAILURES

As mentioned above, all buckling failures were of the same nature. In each case out of plane deformations were initiated by some mechanism. The means of starting the deformations could have been initial warping of the panel, asymmetric loading caused by slight misalignment of the panel in the "picture frame", or asymmetry of the material properties in the layup. This could have been caused by a bias in the direction of weave of the Kevlar/epoxy material, slight misalignment of the patch between the front and rear surfaces, misalignment of the weave direction in applying the patches, or slightly different amounts of adhesive or epoxy between front and back surfaces. Once the out of plane deformation began, the rate of change of the radius of curvature varied widely in the horizontal direction. This radical change in radius of curvature generated a shear in a perpendicular direction to the plane of the panel. While not significantly affecting the Kevlar/epoxy material, this shear had a dramatic effect on the Rohacell foam core material. At the point of shear failure of the foam core, the out of plane rigidity of the entire "mini-sandwich" skin system was drastically reduced, causing large scale buckling in the direction of maximum compression.

It was significant to note that each of the twenty-six panels tested exceeded the limit shear flow requirements, indicating adequate shear flow could be transmitted by similarly sized and oriented panels when mounted onto an aircraft. Only unpatched holes greater than 1.00 inch diameter and the patched holes without filler plugs failed to meet the ultimate shear flow requirement. The out of plane rigidity of the "mini-sandwich" skin system would in fact increase with the curvature of the airframe design. Further criteria must therefore be considered in selection of the most advantageous repair technique.

Ideally, any patch is desired to return the physical and mechanical properties of a damaged item as closely as possible to the original, undamaged properties. Insufficient reinforcement by the patch results in a system which still has a weakness in the damaged and repaired area. Excessive reinforcement of the patch can result in over-stiffening of that area of the skin. Under normal operational loads, resistance of the excessively repaired region to strain imparts additional stress to surrounding regions, potentially causing premature failure of the previously undamaged area.

It was therefore decided that the prebuckling characteristics of the repair most closely conform to the results obtained from the undamaged panels. In this regard, the slopes and the buckling loads of the maximum shear strain versus applied load curves were compared with respect to the variations in the four parameters.

Hole size was found to have a profound impact on the test results. Repairs of smaller sized holes were found to conform more closely to the undamaged panels than repairs of larger sized holes.

The amount of patch overlap had a significant effect as well, and tended to support the premise stated earlier of closely matching the patch to the surrounding material. In each case, the 0.50 inch overlap performed better than the 0.25 inch overlap of the same type repair. Likewise, the 0.50 inch overlap performed better than the 0.75 inch overlap. This would suggest not only that an overlap of 0.75 inches was excessive for the repair needed, but also that the best overlap size is largely independent of the hole diameter for the size repairs investigated.

The selection of a filler material for the plug also had a marked effect. Clearly the use of no plug filler (Panels N, O, and P) resulted in significantly less performance than the use of filler material. The properties of panels with structural adhesive plugs were slightly more degraded than the properties of panels with Rohacell foam plugs, though the differences were not of great proportion. The complication of delamination of the patch over the plug itself, however, remains. Additionally, use of the fluid structural adhesive would be limited to components which were either horizontally oriented or could be removed from the aircraft. Otherwise the adhesive would tend to flow out of the patch region under the effect of gravity.

The selection of a patch material had the significance of demonstrating that properties of panels repaired with fiberglass patches did not vary appreciably from those of panels repaired with Kevlar/epoxy patches.

Figure 17 shows a summary of the limit shear flow carrying capacity for each patched panel, based on hole size. Figure 18 summarizes the ultimate shear flow carying capacity of the unrepaired holes as dependent upon hole size.

C. ASSESSMENT OF POSTBUCKLING TESTS

The inclusion of applied load versus total head displacement curves in Appendix C was intended to qualitatively demonstrate the reduction in the energy absorbing capability of panels with increasingly larger holes cut from their centers. Additionally, it provides a source of test data in the postbuckled regime for energy analysis of "mini-sandwich" construction with Kevlar/epoxy and Rohacell

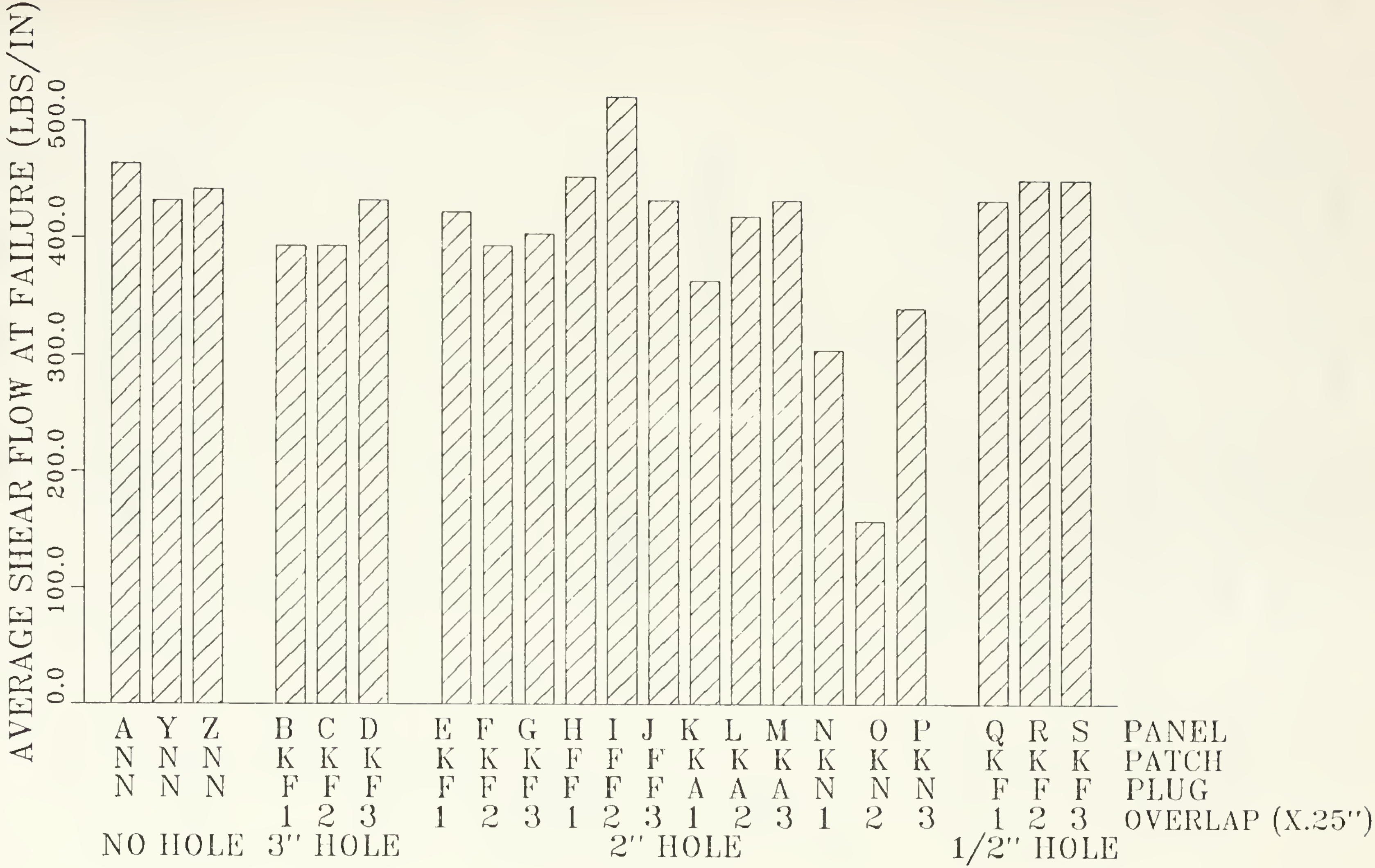


Fig. 17. Summary of Patch Results

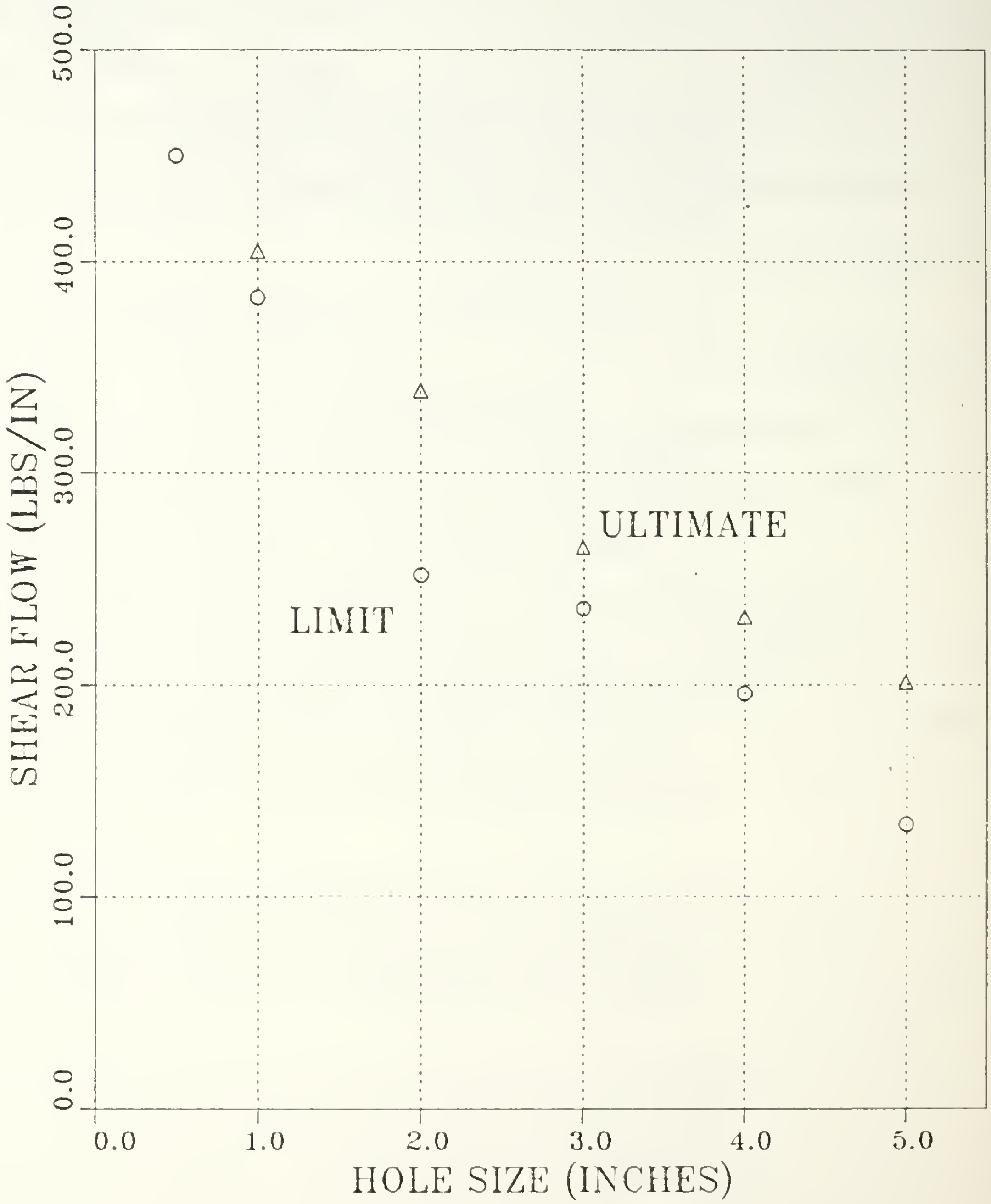


Fig. 18. Effect of Hole Size on Shear Flow Carrying Capacity

foam. The horizontal cracking of each of the panels with holes was restricted to the test area, eliminating any stress concentrations in the "picture frame" or loading tabs from consideration as sources for the cracking. Ignoring any data taken after failure induced by through bolt holes in Panels Y and Z insures data was not skewed significantly by flaws inherent in the test procedure. Rather, it portrays a more accurate picture of the energy absorbing capability of the "mini-sandwich" skin.

VII. CONCLUSIONS

Whereas both Kevlar/epoxy and fiberglass patches applied over Rohacell foam plugs yielded approximately the same results, the use of Kevlar/epoxy patches by a field maintenance activity has serious drawbacks. It necessitates that a method for curing the Kevlar/epoxy patches under temperature and pressure be available. While use of heating elements and pads would achieve the requisite temperatures for curing, a practical method for maintaining pressure on the patch is lacking. Even if a portable press were developed, it would be necessary to have facings manufactured for the press to match every contour on the airframe. It is unrealistic for a field maintenance activity to support such a repair method.

Fiberglass patches, on the other hand, provide considerable advantages. Fiberglass fabric is easily shaped to fit any contour that would be encountered on the airframe. EA-956 structural adhesive has a pot life of approximately thirty minutes at room temperature, and completely cures at 150-200 degrees Fahrenheit in one hour (at room temperature it reaches 75 percent of its maximum strength in twenty-four hours), making it a good choice for use in field repairs. Fiberglass and EA-956 structural adhesive are inexpensive, and readily available, and require no special storage requirements except for prolonged periods in areas of elevated temperatures. Rohacell 71WF also is easily handled and stored, readily available, and fairly inexpensive.

For the above mentioned reasons, the recommended field repair technique for holes up to five inches in diameter is outlined below.

Step 1: Cut smooth, circular hole around damaged area.

Step 2: Recess foam core 0.125 inches around circumference of hole.

Step 3: Cut circular disk of Rohacell 71WF material, and sand to fit tightly into hole.

Step 4: Clean disk and hole.

Step 5: Fill recessed area with EA-956 structural adhesive.

Step 6: Insert disk into hole and apply heat from hand held heating gun until adhesive hardens (about ten minutes).

Step 7: Sand both sides of disk even with surrounding skin and clean.

Step 8: Cut circular piece of fiberglass fabric with its radius 1/2 inch larger than hole radius.

Step 9: Apply thin coat of EA-956 to one side of plug and area within 1/2 inch of plug.

Step 10: Center fiberglass material over foam plug and tamp lightly to saturate completely. Insure to align weave pattern of patch to surrounding material.

Step 11: Apply EA-956 as required to saturate fiberglass fabric and create smooth surface.

Step 12: Apply heat until hardened (about ten minutes).

Step 13: Repeat Steps 8 through 12 for remaining side.

VIII. RECOMMENDATIONS

A finite element analysis should be performed to examine the effectiveness of finite element models to predict buckling failures of laminated sturctures where failure criteria of included lamina vary greatly, as in the case of "mini-sandwich" material.

Conduct fatigue testing of the various repair methods to see if significant differences from static shear testing result.

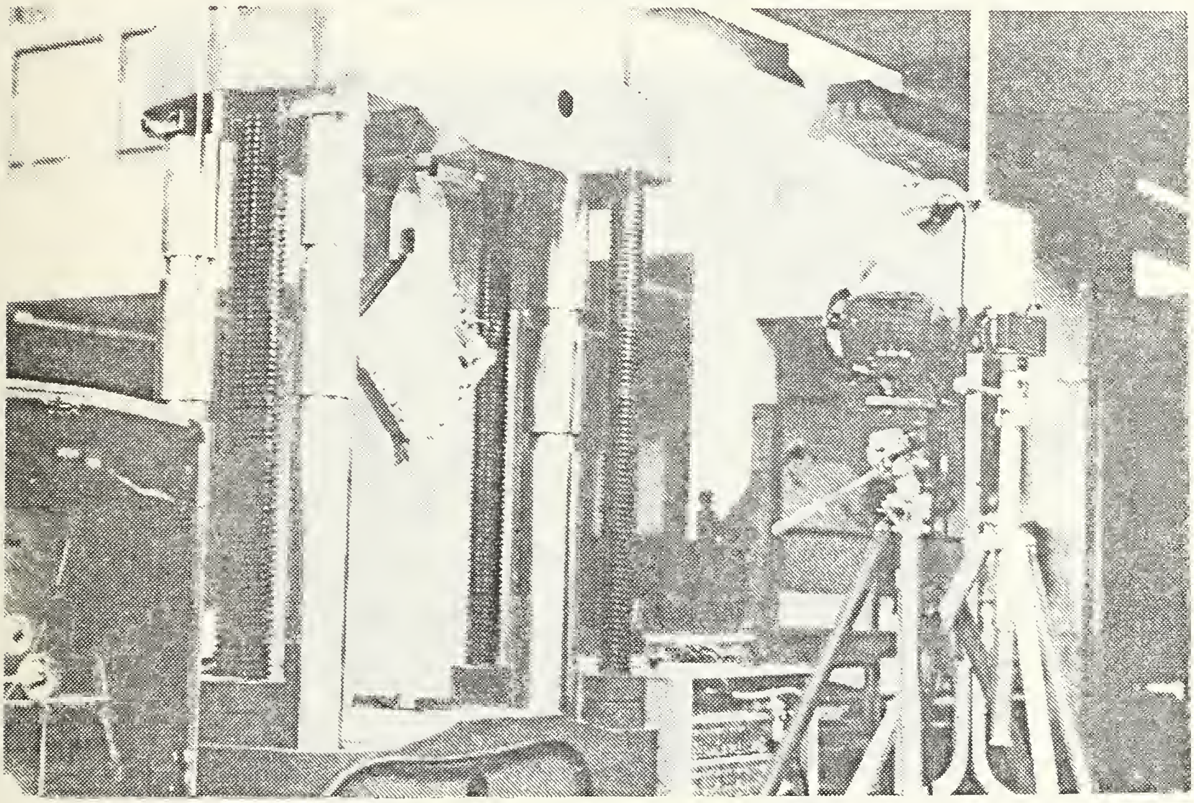
Conduct environmental tests to determine the effects of humidity, temperature cycling, ultra-violet radiation and corrosive agents on various repair techniques.

Design and construct larger shear testing fixture to investigate the effects of holes of larger size than five inch diameter.

APPENDIX A

PHOTOGRAPHS OF PHOTOELASTIC TEST PANEL

This appendix contains photographs of a test panel constructed of photoelastic material and loaded incrementally until failure. The process used for taking the photographs was a reflective process. A polarized light was shown against the panel, which had been coated on the opposite side with reflective aluminized paint. The reflected light was photographed through a polariscope, as depicted below. The results for successive loads are shown in the following pages.

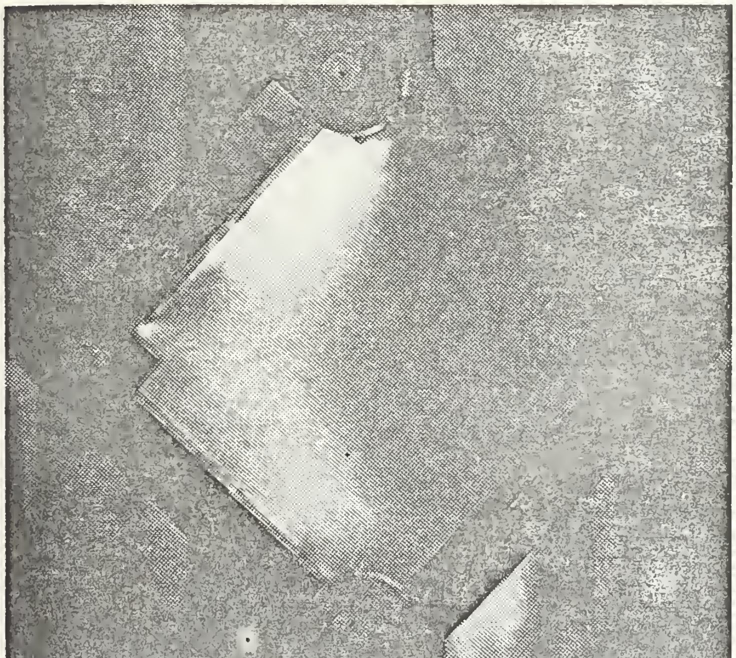


Apparatus for Photographing Photoelastic Test Panel

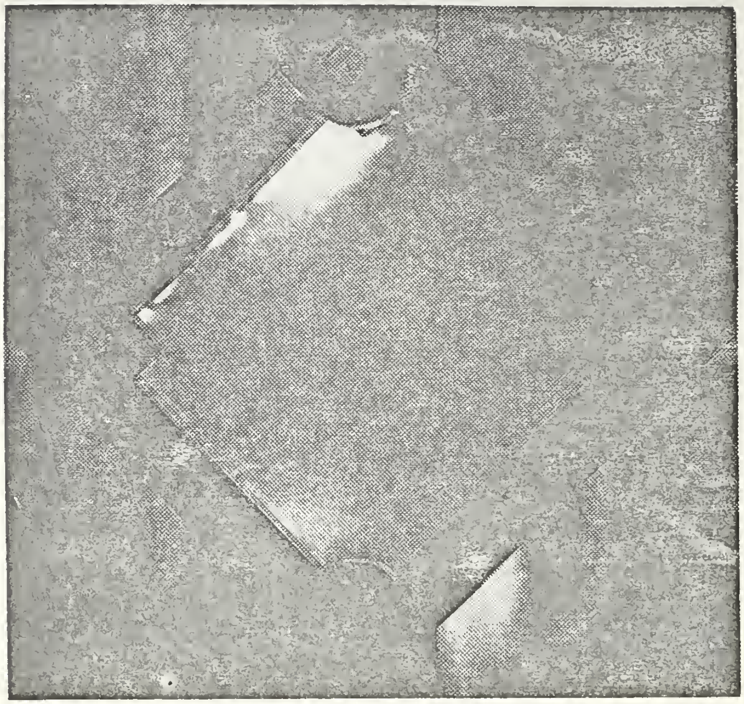
50 Pounds Applied Load

100 Pounds Applied Load

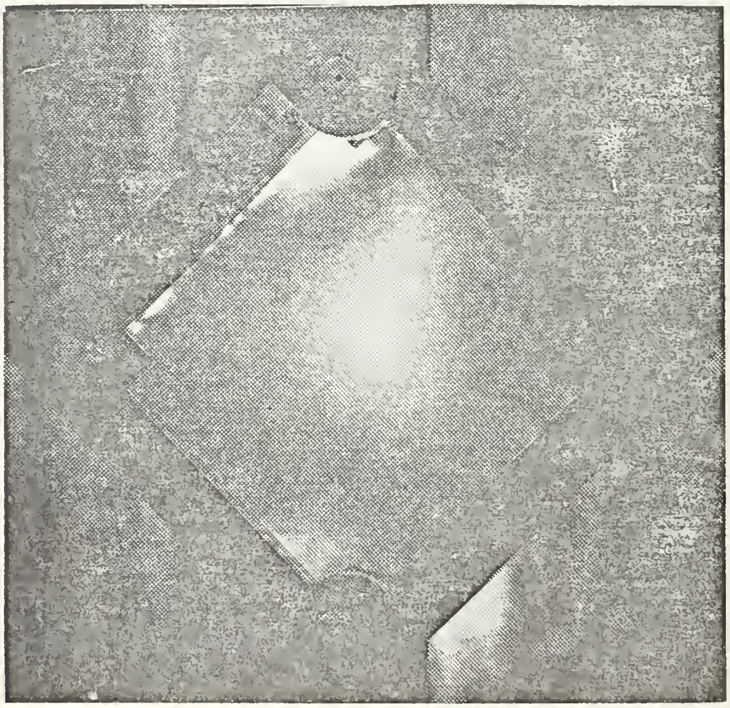
150 Pounds Applied Load



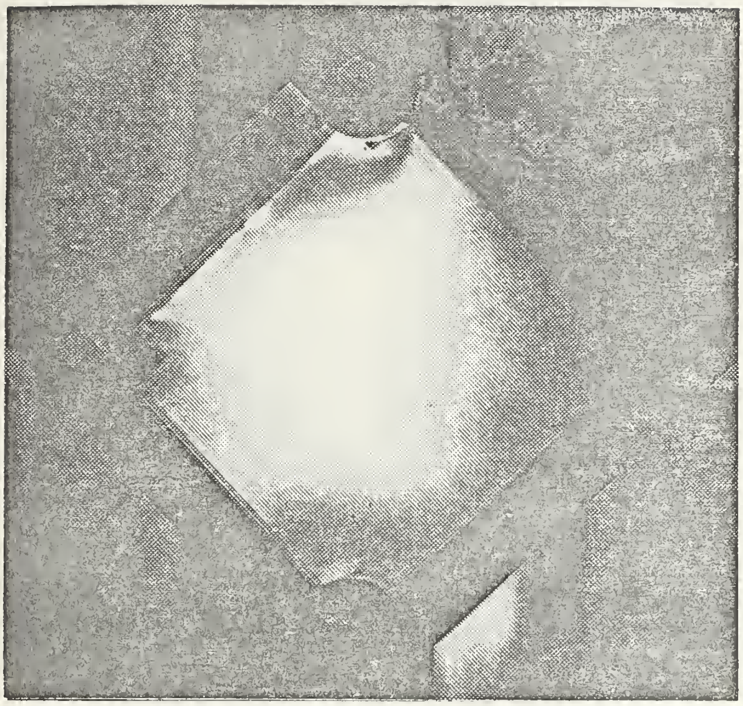
200 Pounds Applied Load



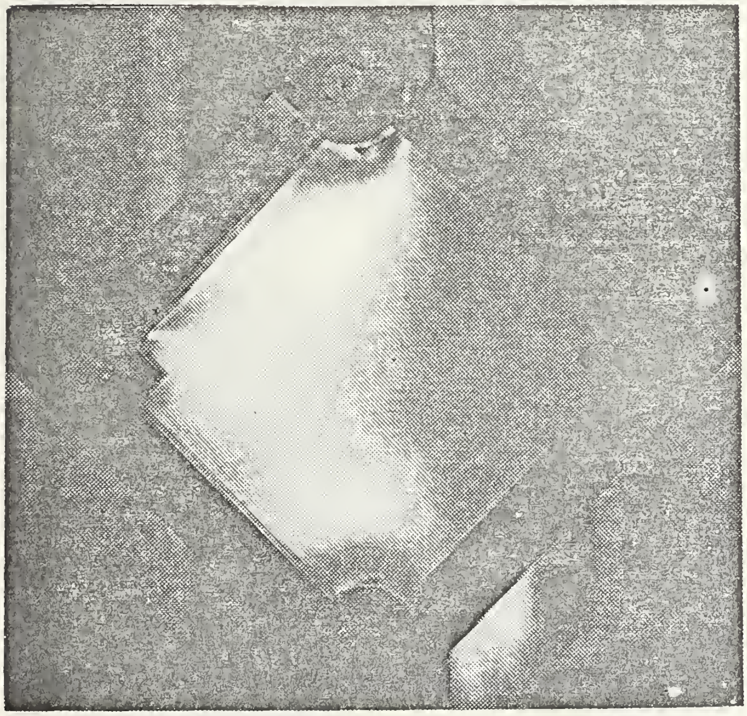
250 Pounds Applied Load



300 Pounds Applied Load



400 Pounds Applied Load



500 Pounds Applied Load

750 Pounds Applied Load

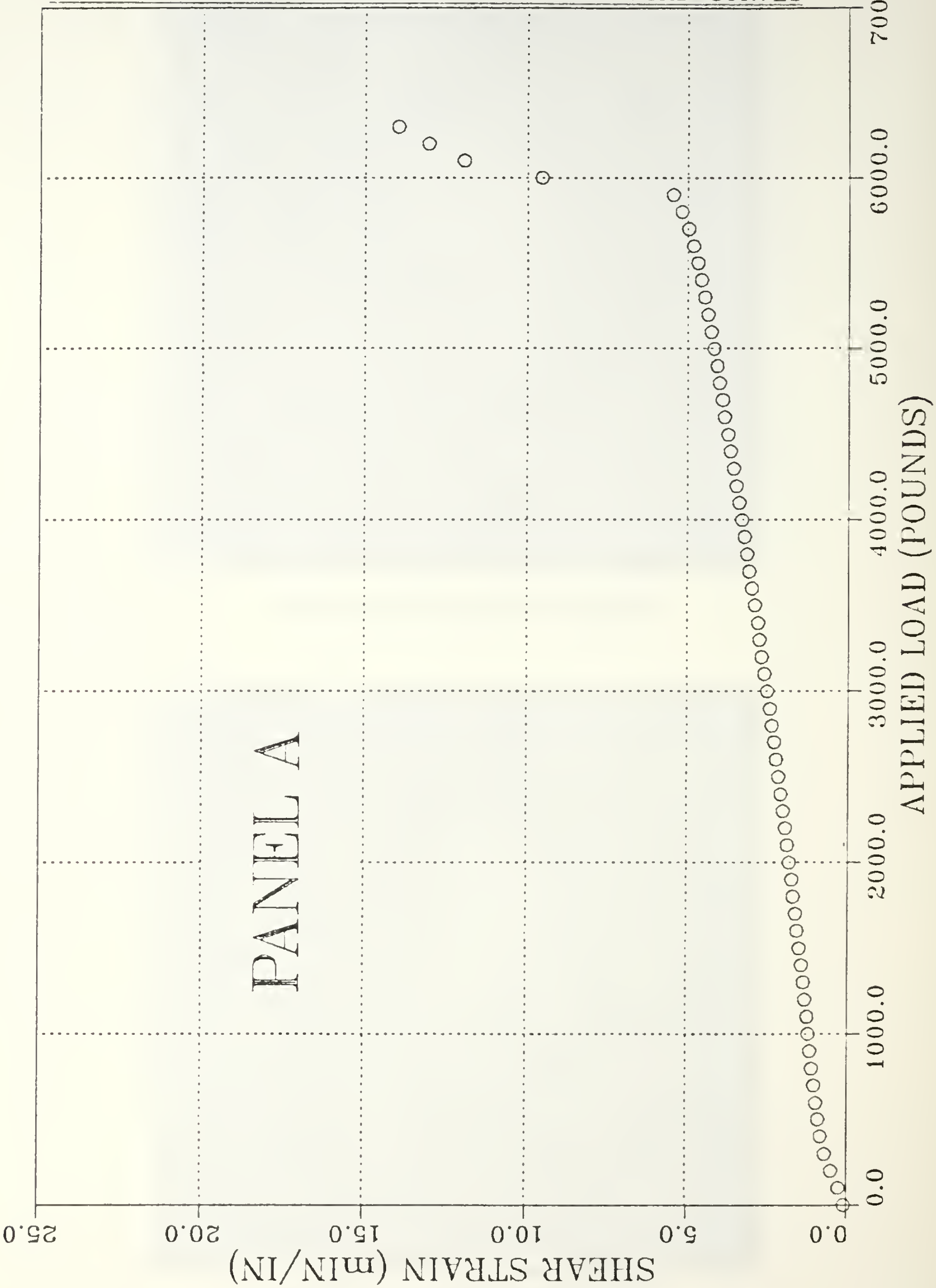
1000 Pounds Applied Load

1400 Pounds Applied Load

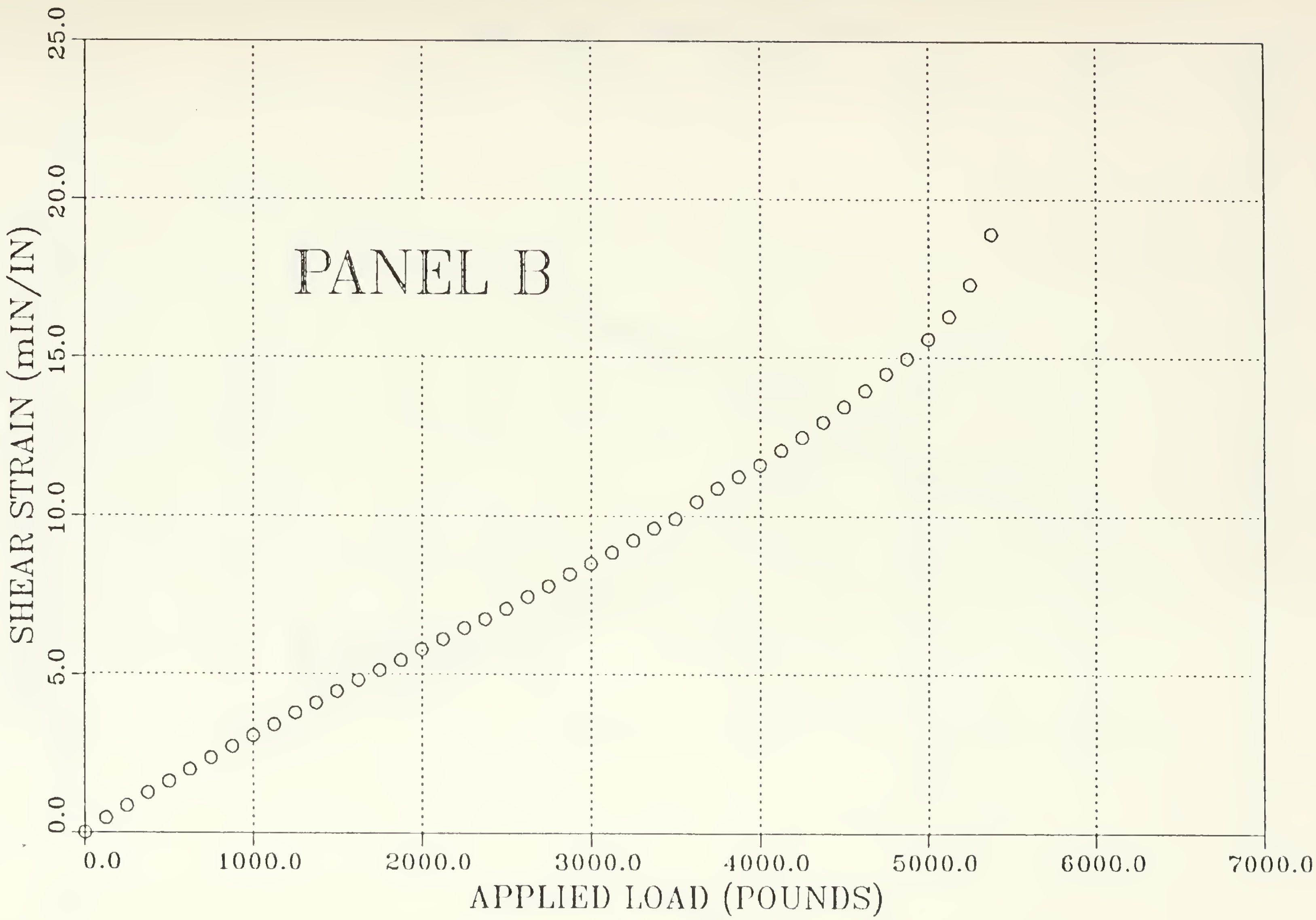
1800 Pounds Applied Load

APPENDIX B

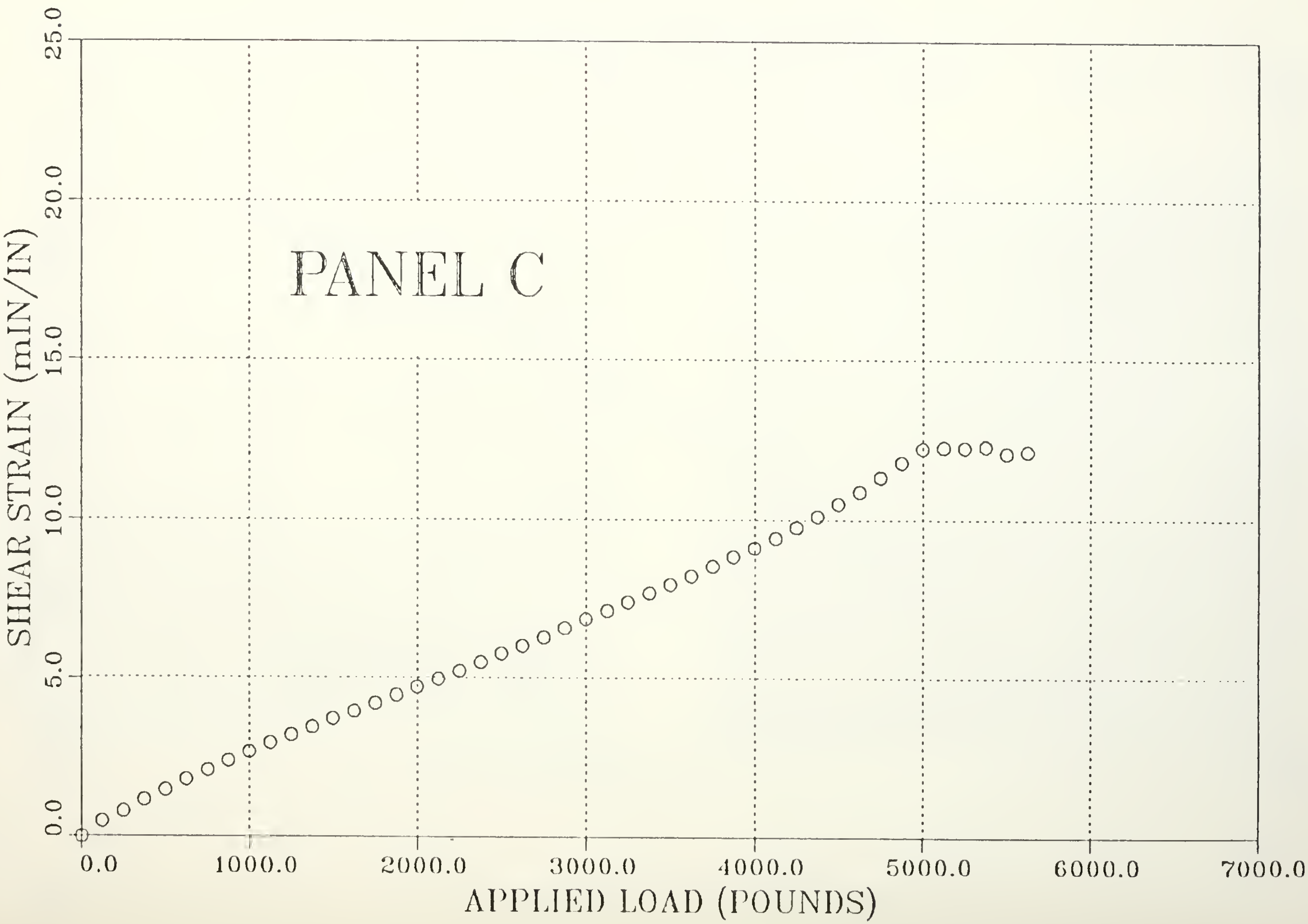
MAXIMUM SHEAR STRAIN VERSUS APPLIED LOAD CURVES



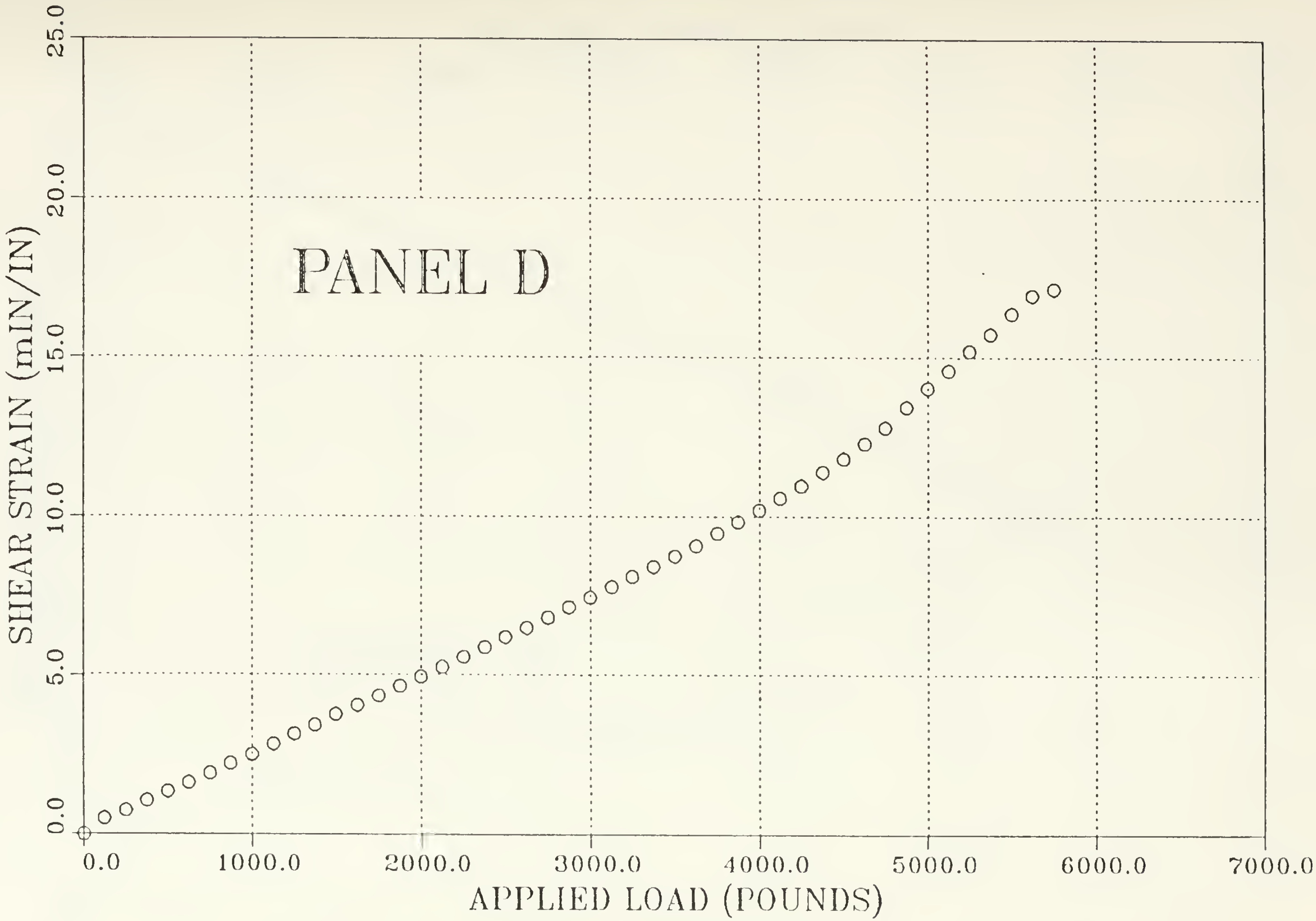
Panel A - Solid Panel



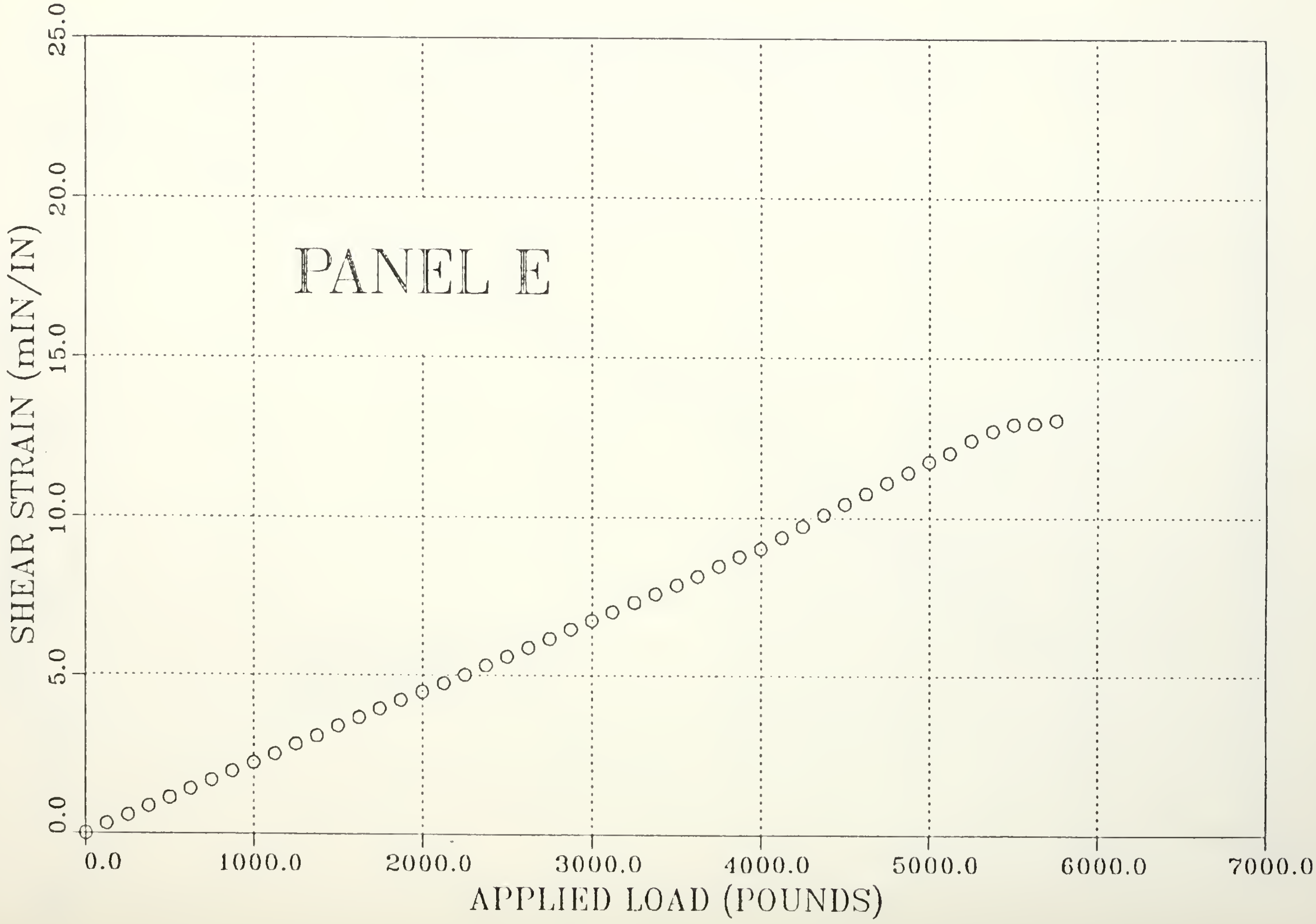
Panel B - 3" Hole, Kevlar Patch, Foam Plug, 1/4" Overlap



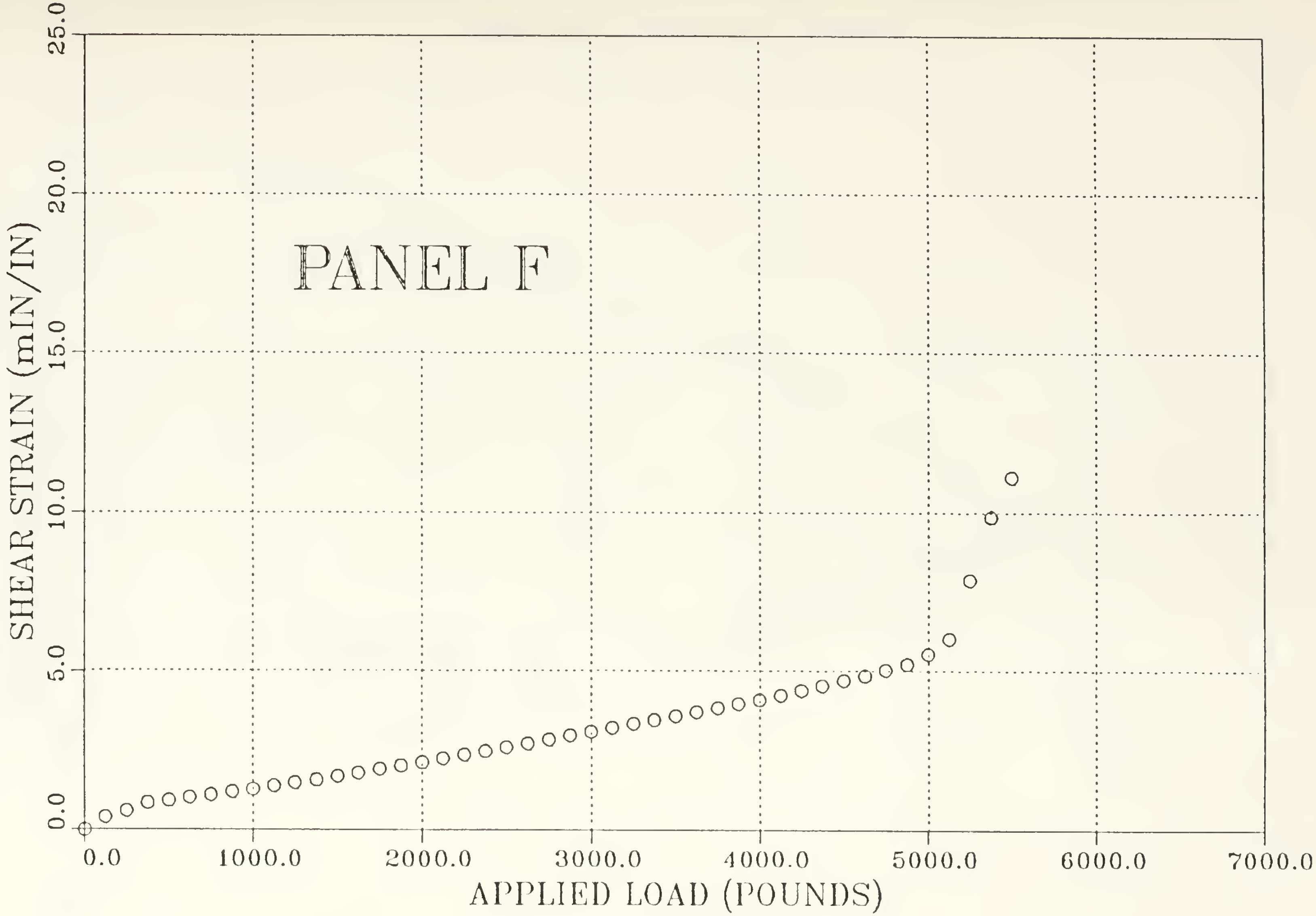
Panel C - 3" Hole, Kevlar Patch, Foam Plug, 1/2" Overlap



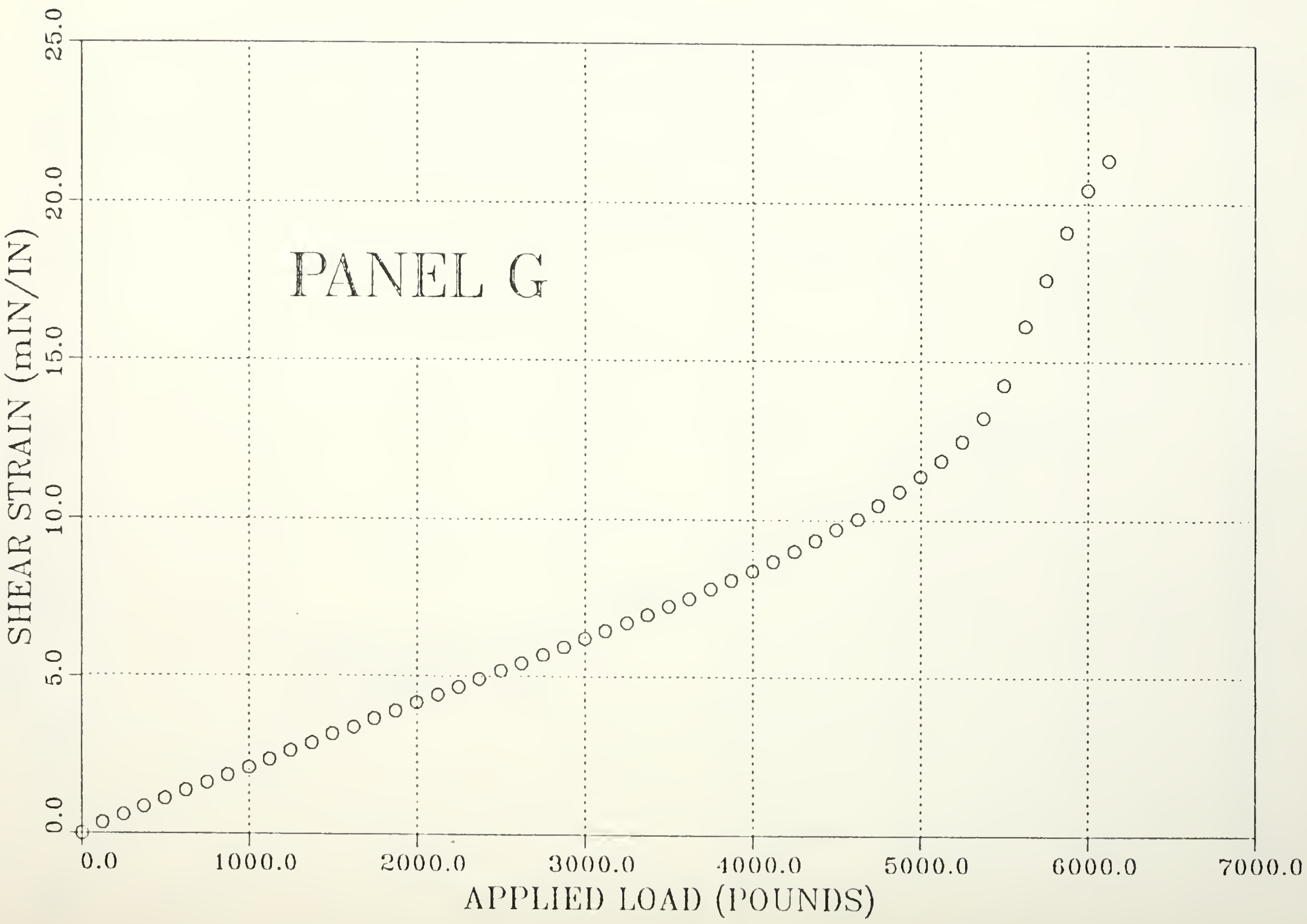
Panel D - 3" Hole, Kevlar Patch, Foam Plug, 3/4" Overlap



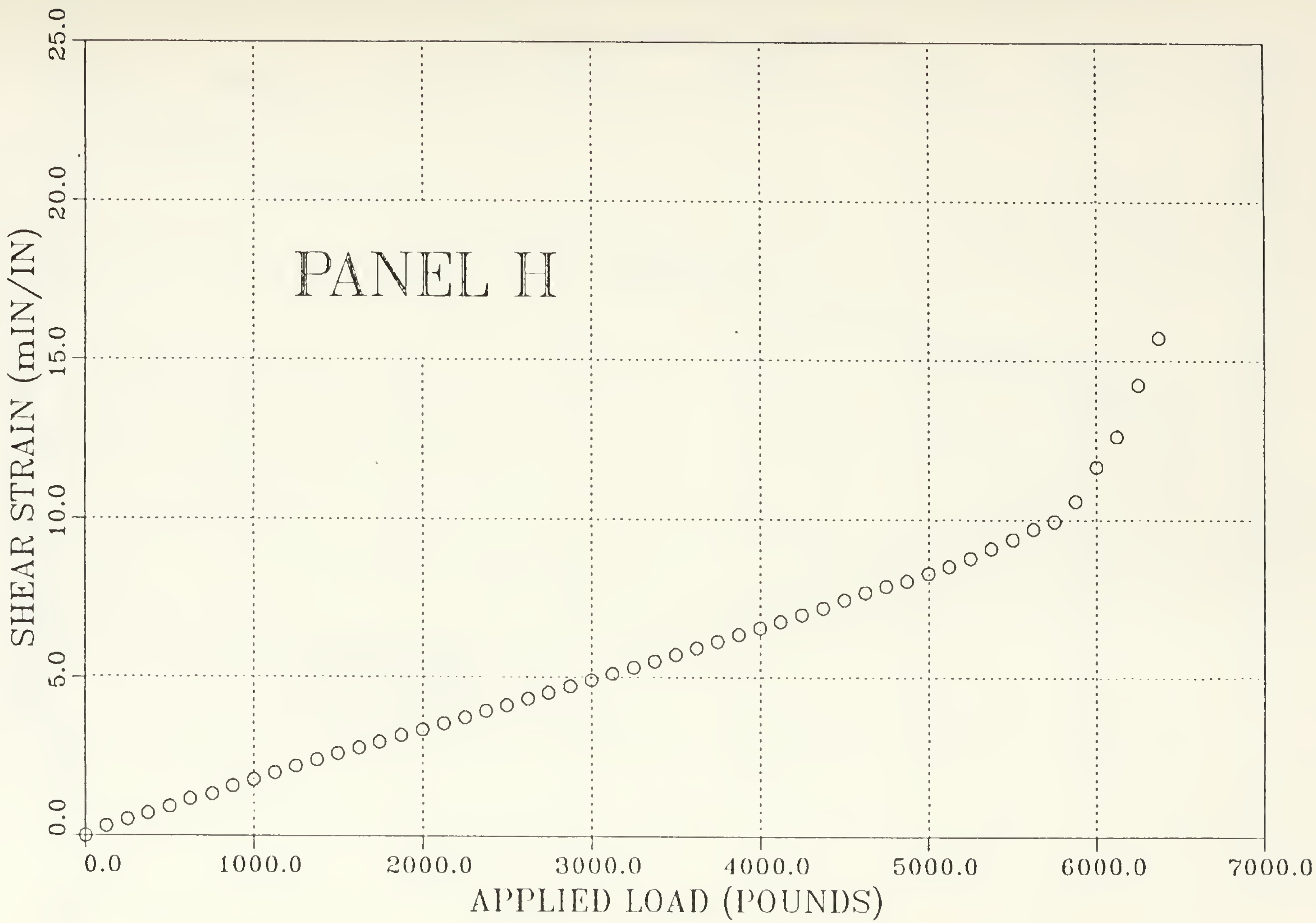
Panel E - 2" Hole, Kevlar Patch, Foam Plug, 1/4" Overlap



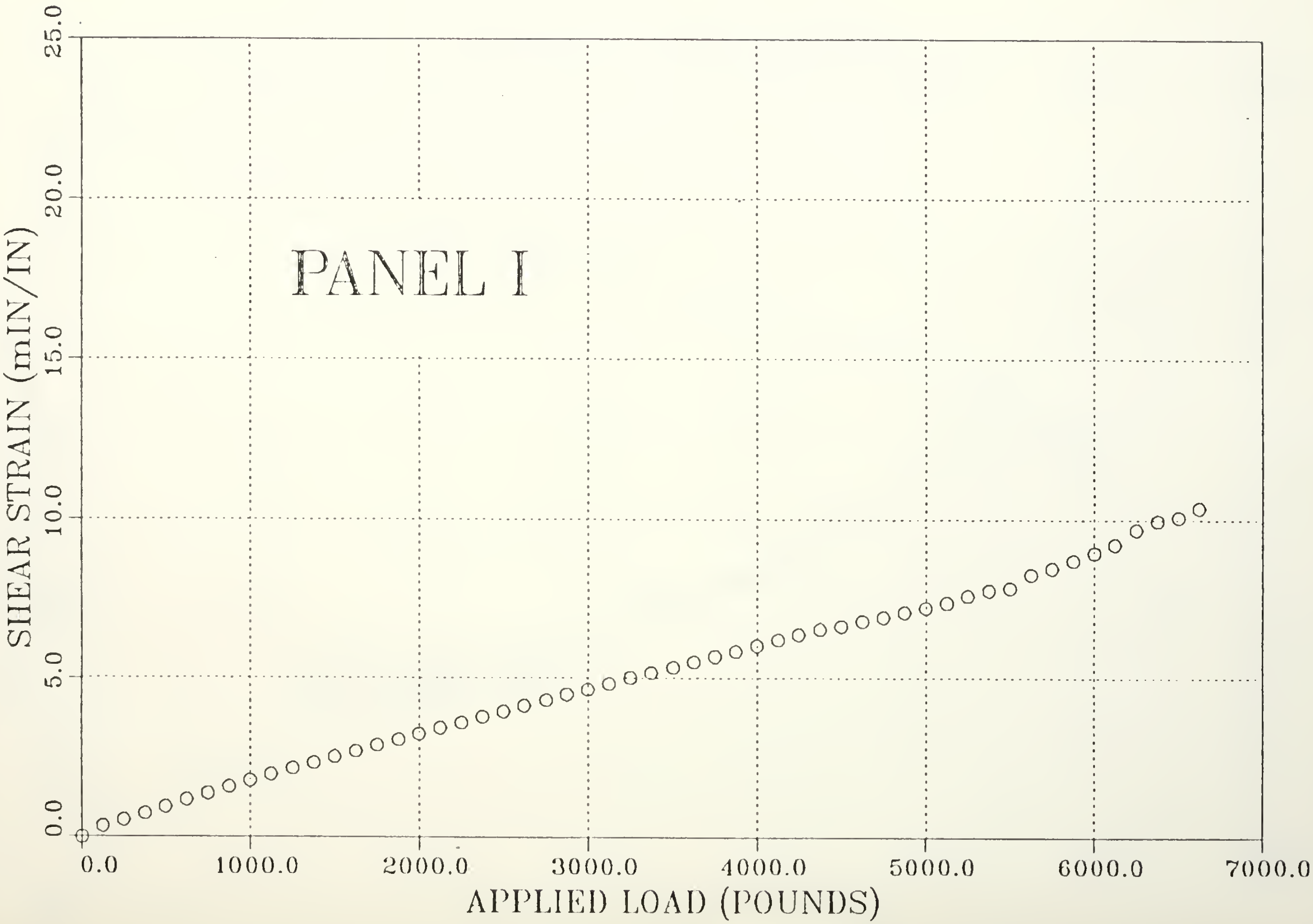
Panel F - 2" Hole, Kevlar Patch, Foam Plug, 1/2" Overlap



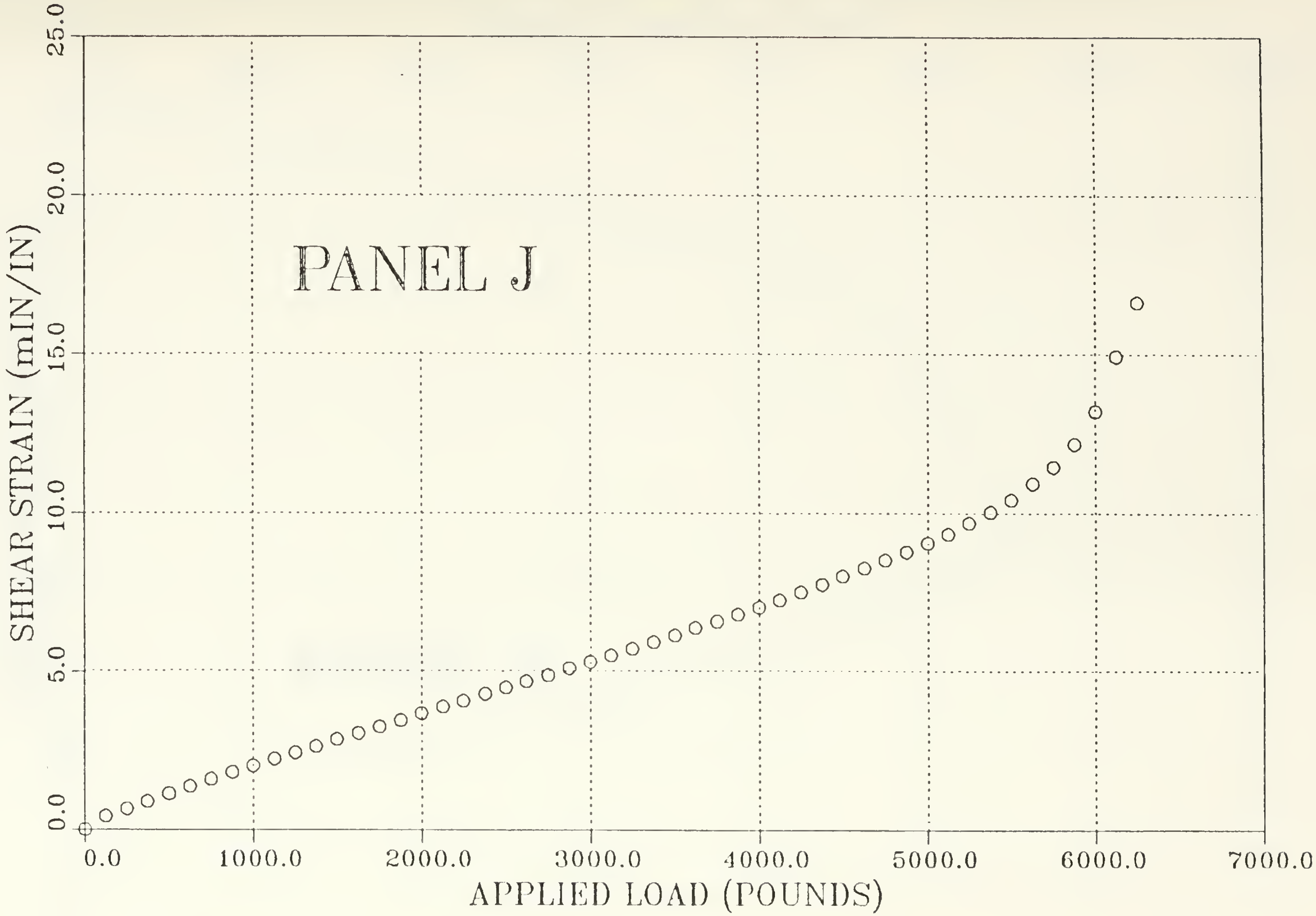
Panel G - 2" Hole, Kevlar Patch, Foam Plug, 3/4" Overlap



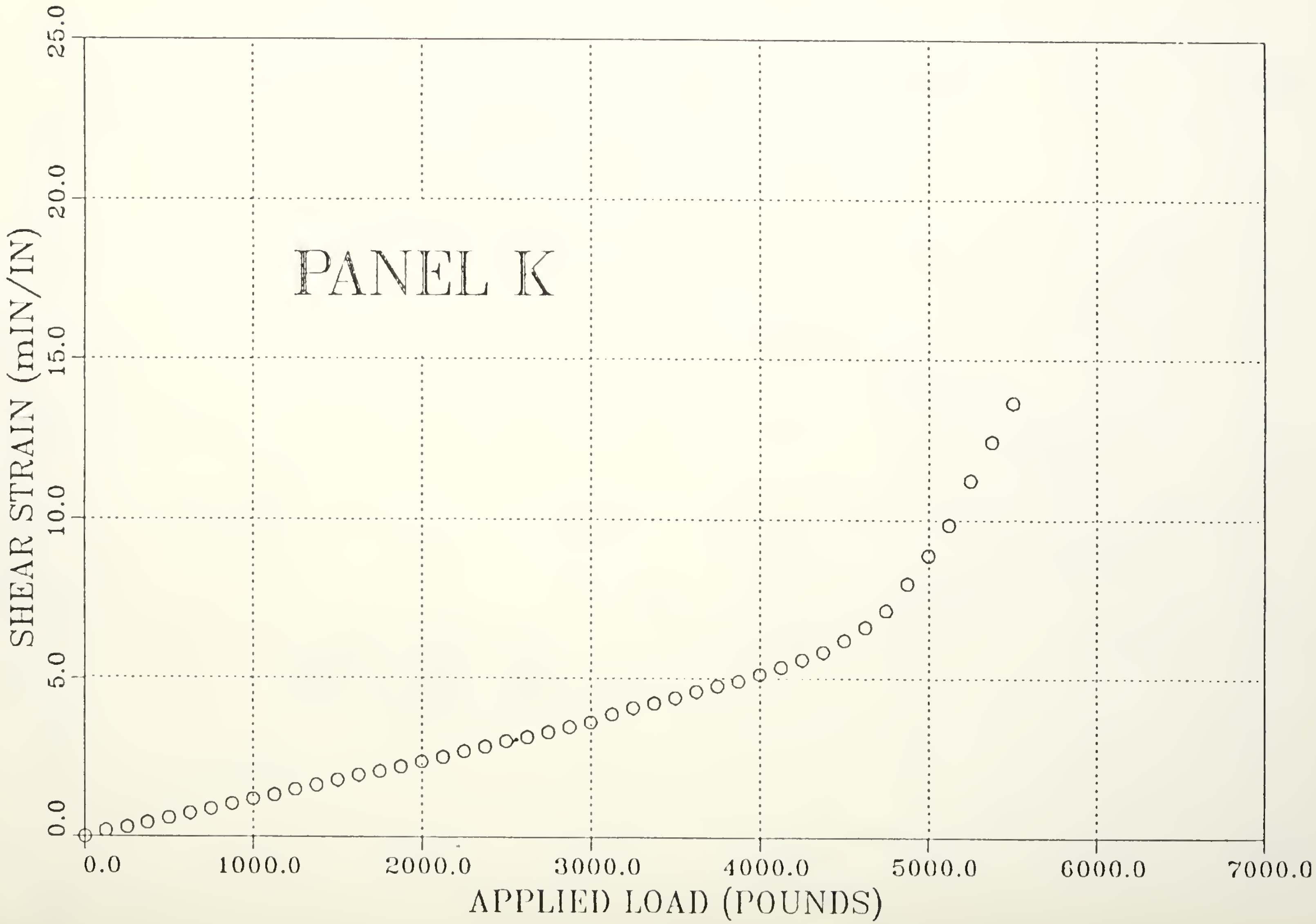
Panel H - 2" Hole, Fiberglass Patch, Foam Plug, 1/4" Overlap



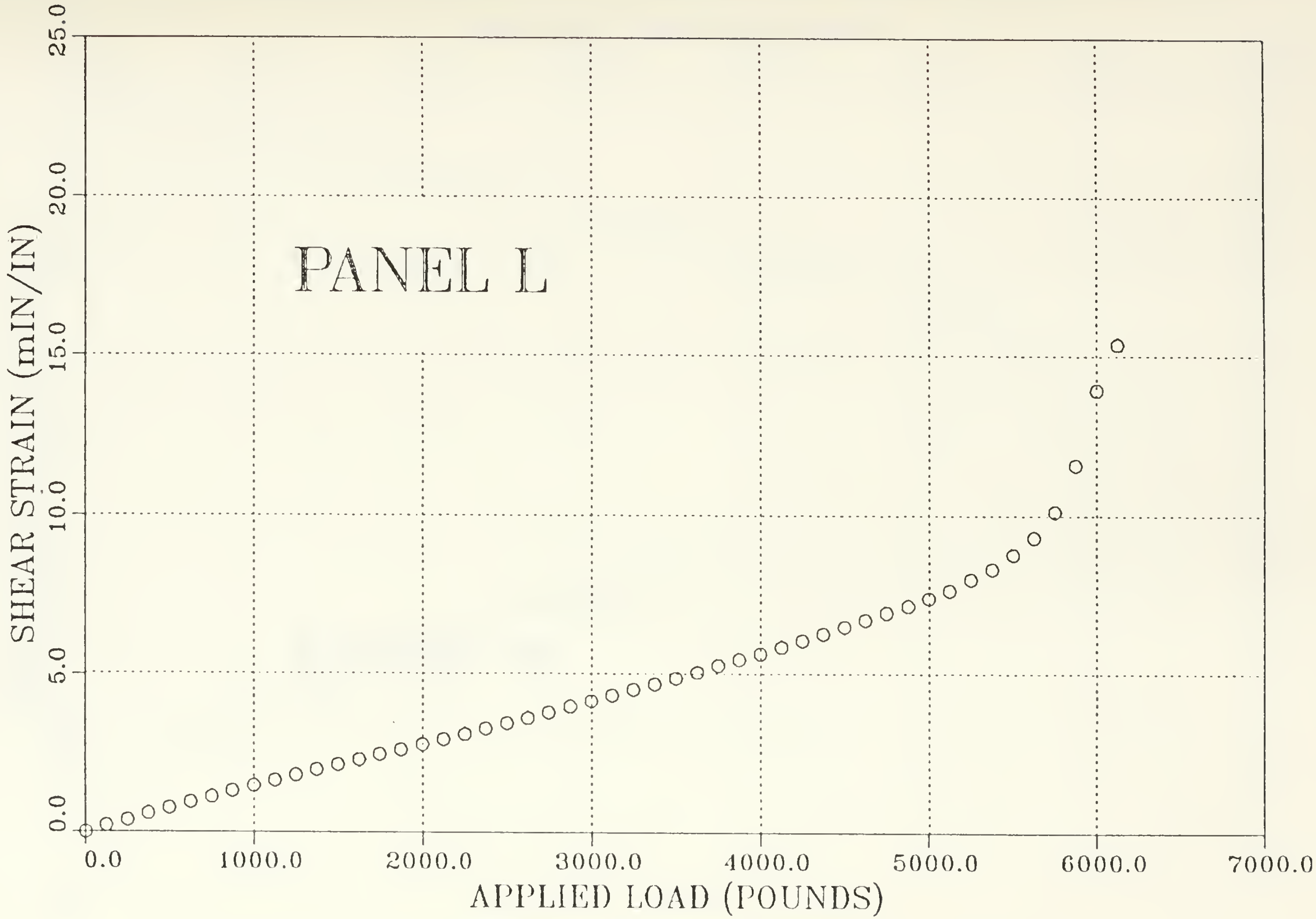
Panel I - 2" Hole, Fiberglass Patch, Foam Plug, 1/2" Overlap



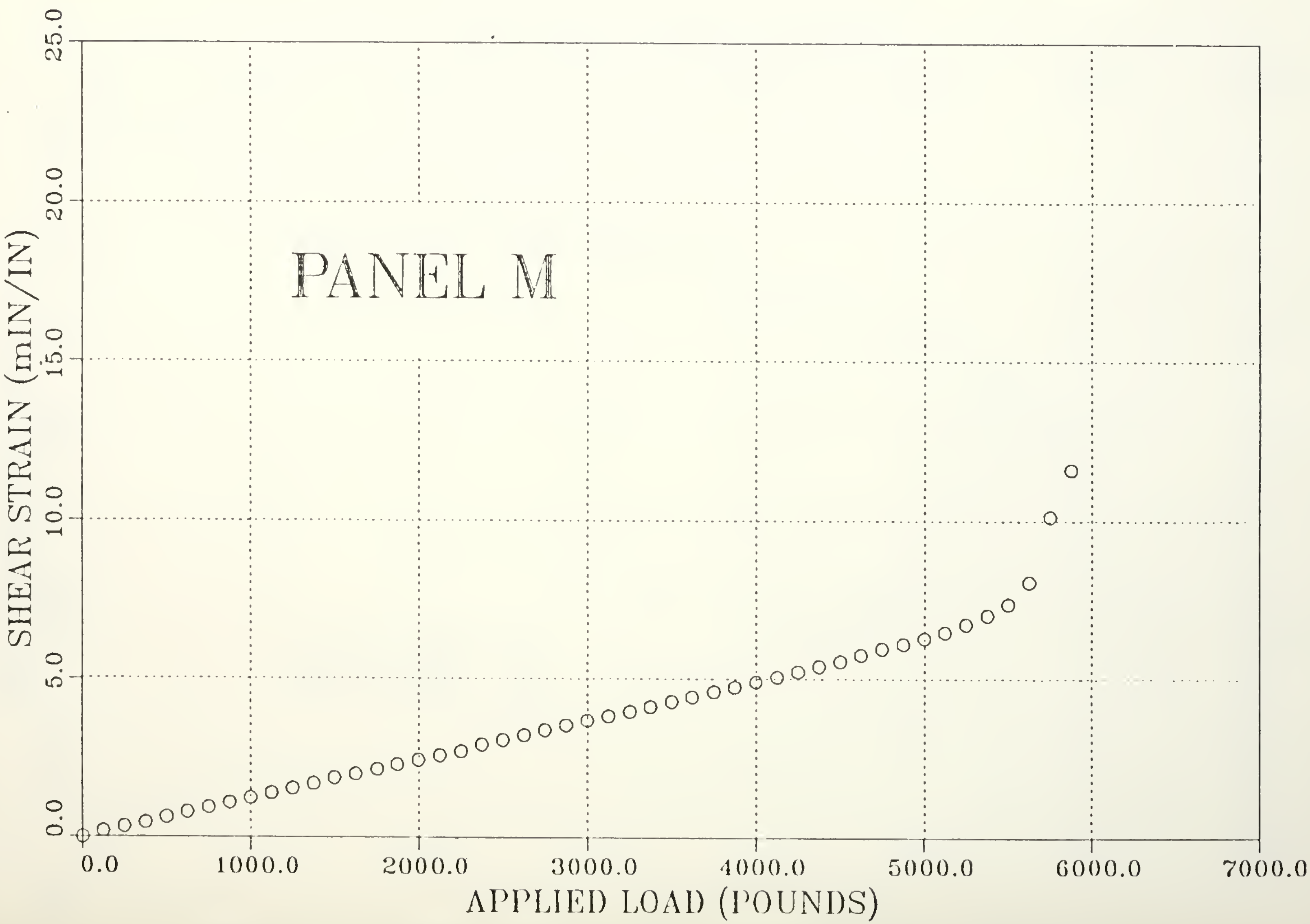
Panel J - 2" Hole, Fiberglass Patch, Foam Plug, 3/4" Overlap



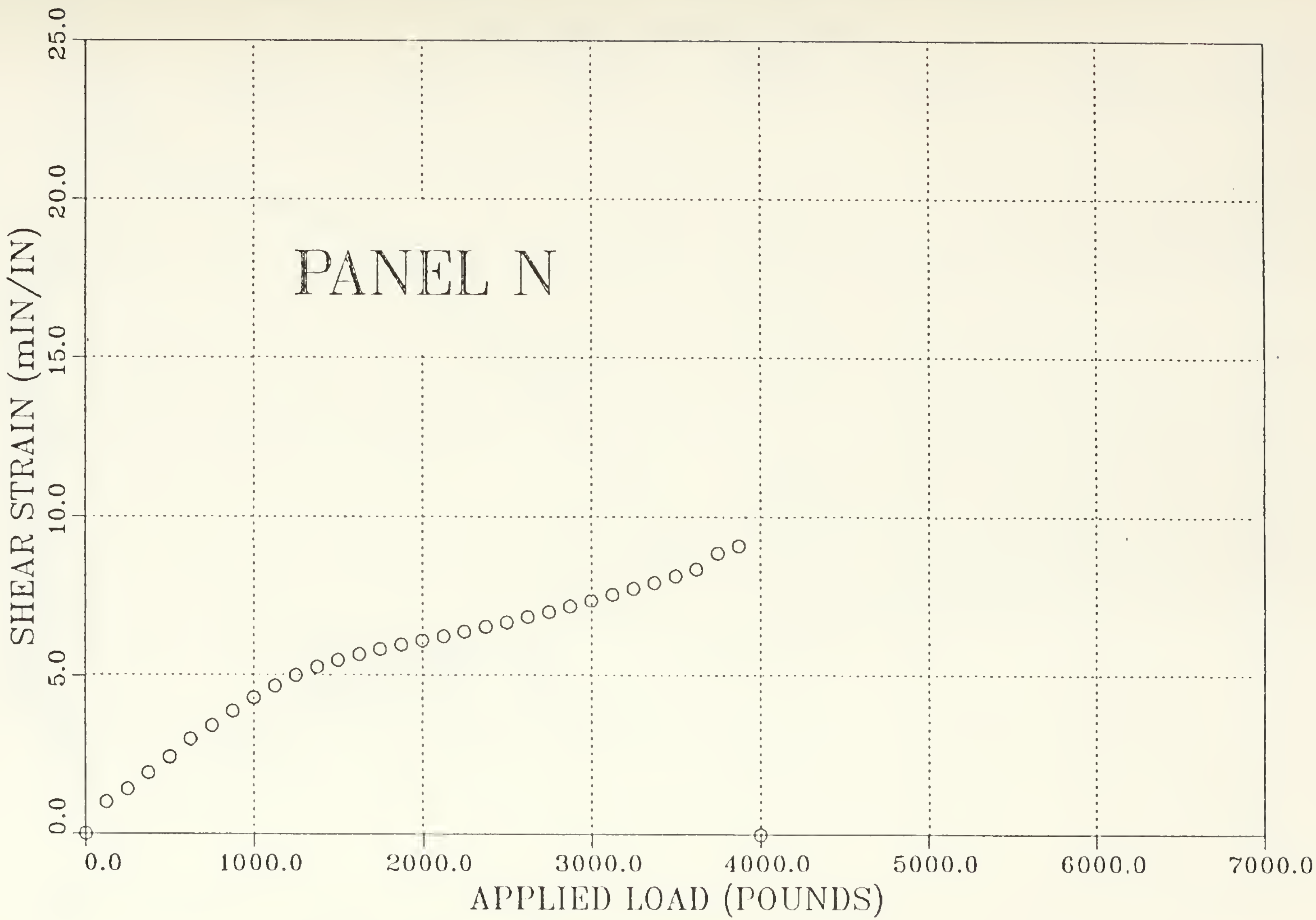
Panel K - 2" Hole, Kevlar Patch, Adhesive Plug, 1/4" Overlap



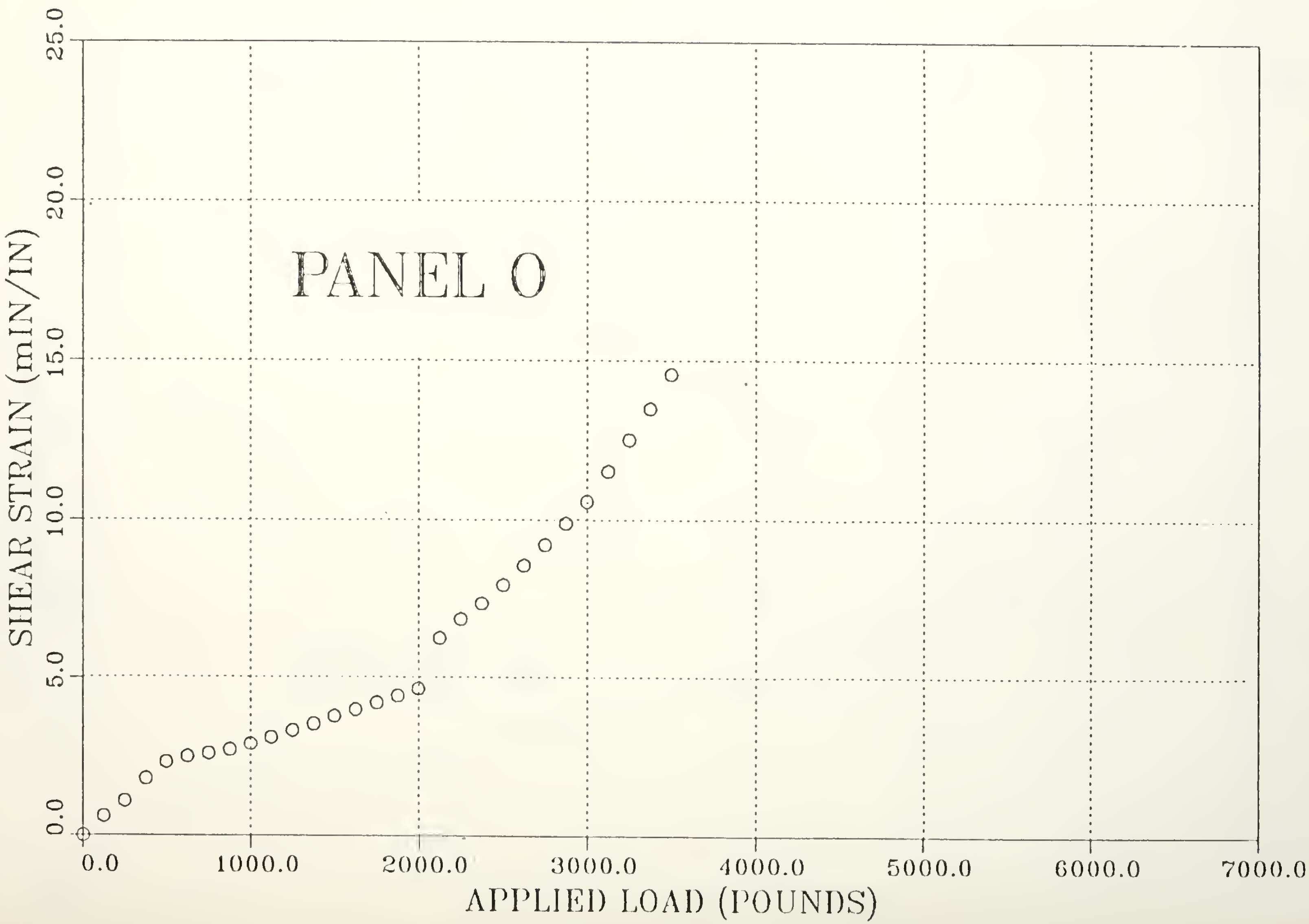
Panel L - 2" Hole, Kevlar Patch, Adhesive Plug, 1/2" Overlap



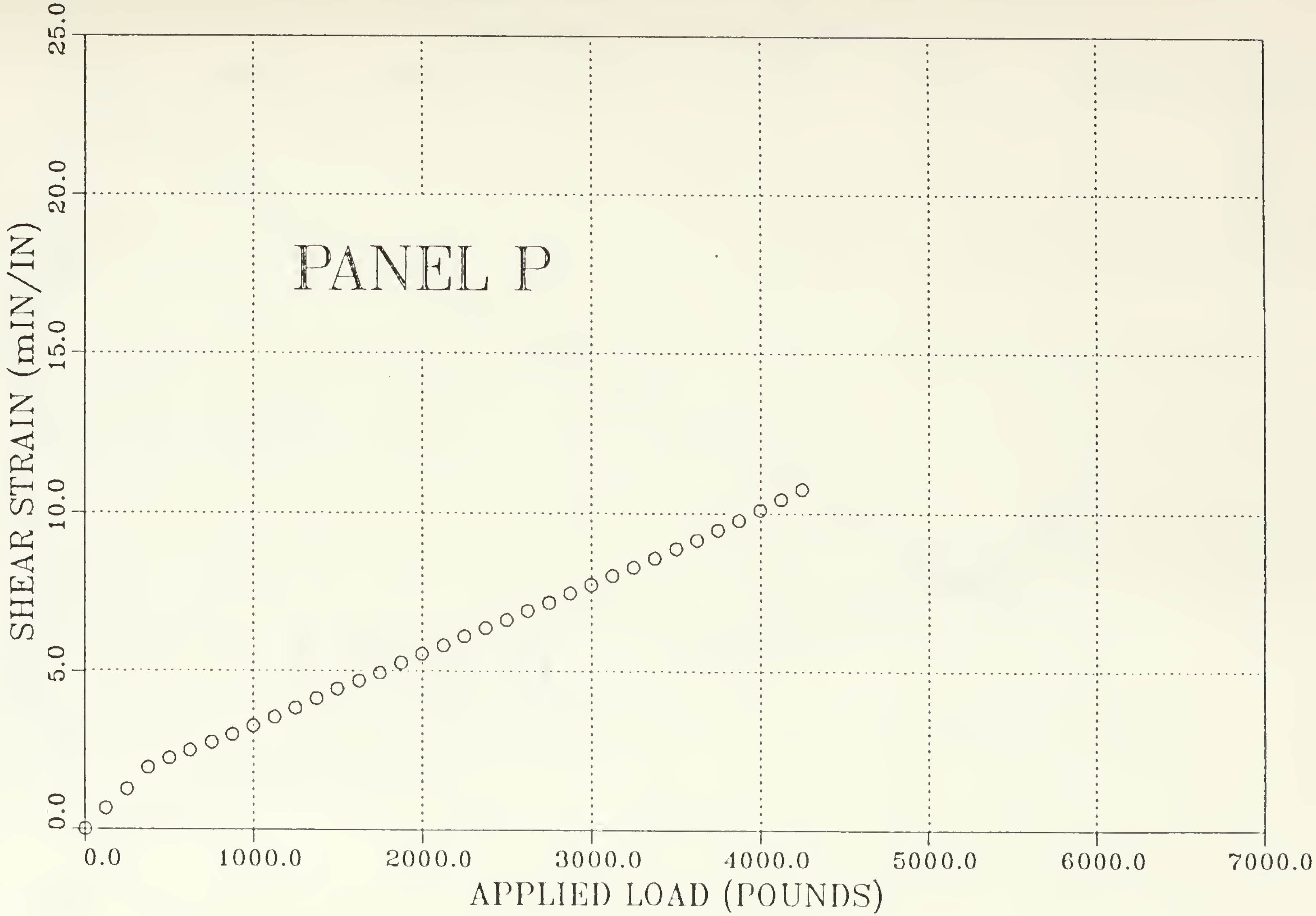
Panel M - 2" Hole, Kevlar Patch, Adhesive Plug, 3/4" Overlap



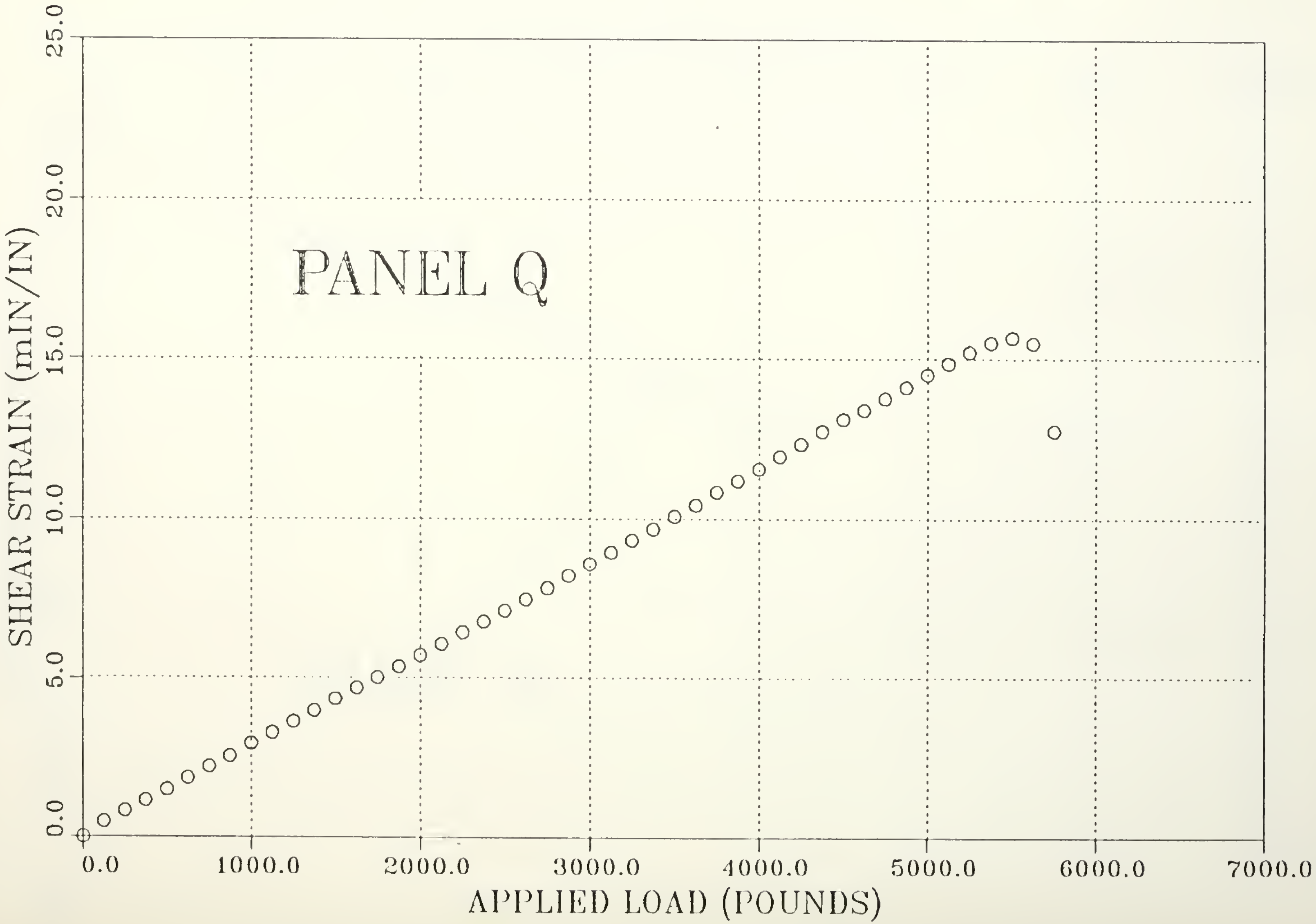
Panel N - 2" Hole, Kevlar Patch, No Plug, 1/4" Overlap



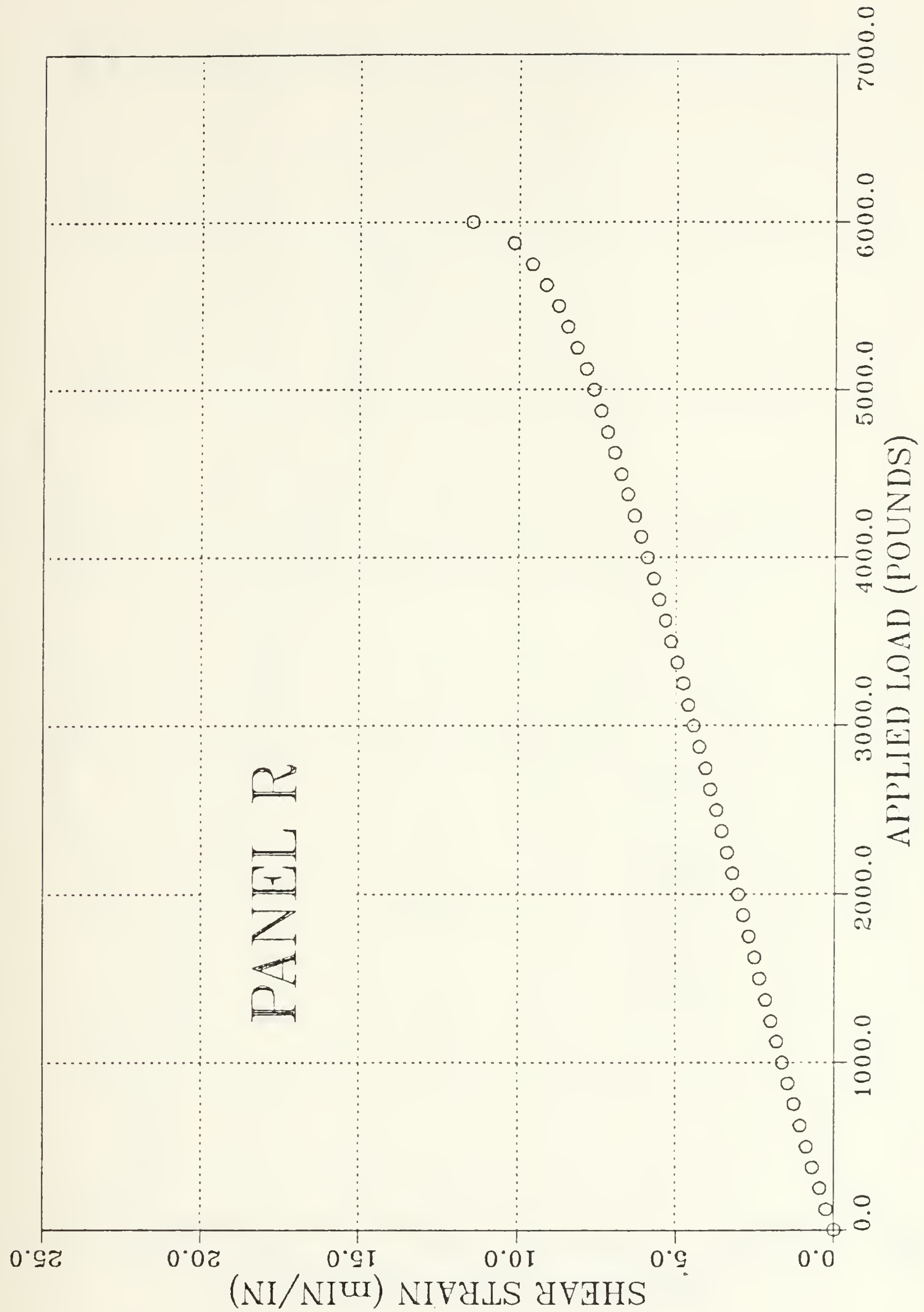
Panel O - 2" Hole, Kevlar Patch, No Plug, 1/2" Overlap



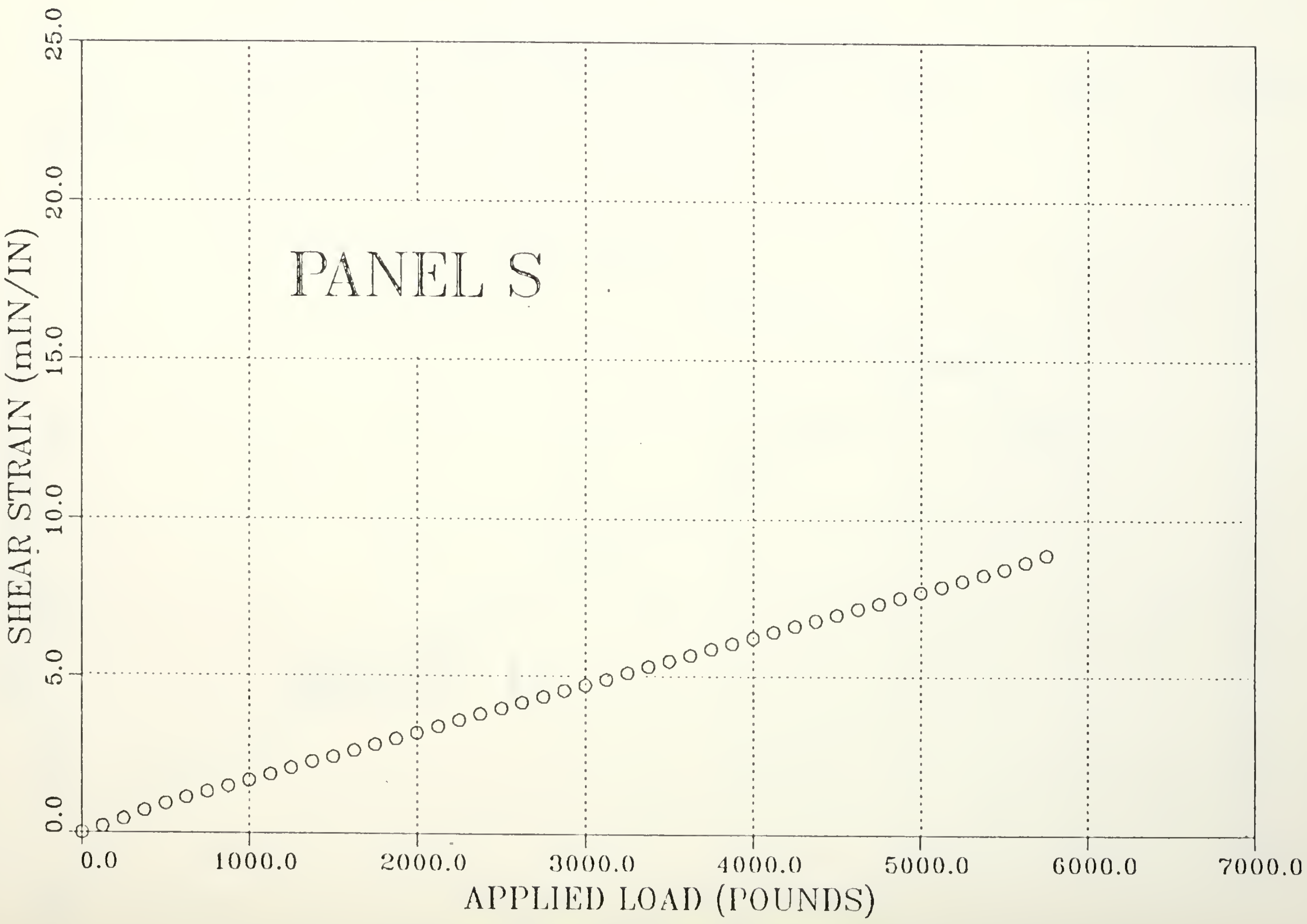
Panel P - 2" Hole, Kevlar Patch, No Plug, 3/4" Overlap



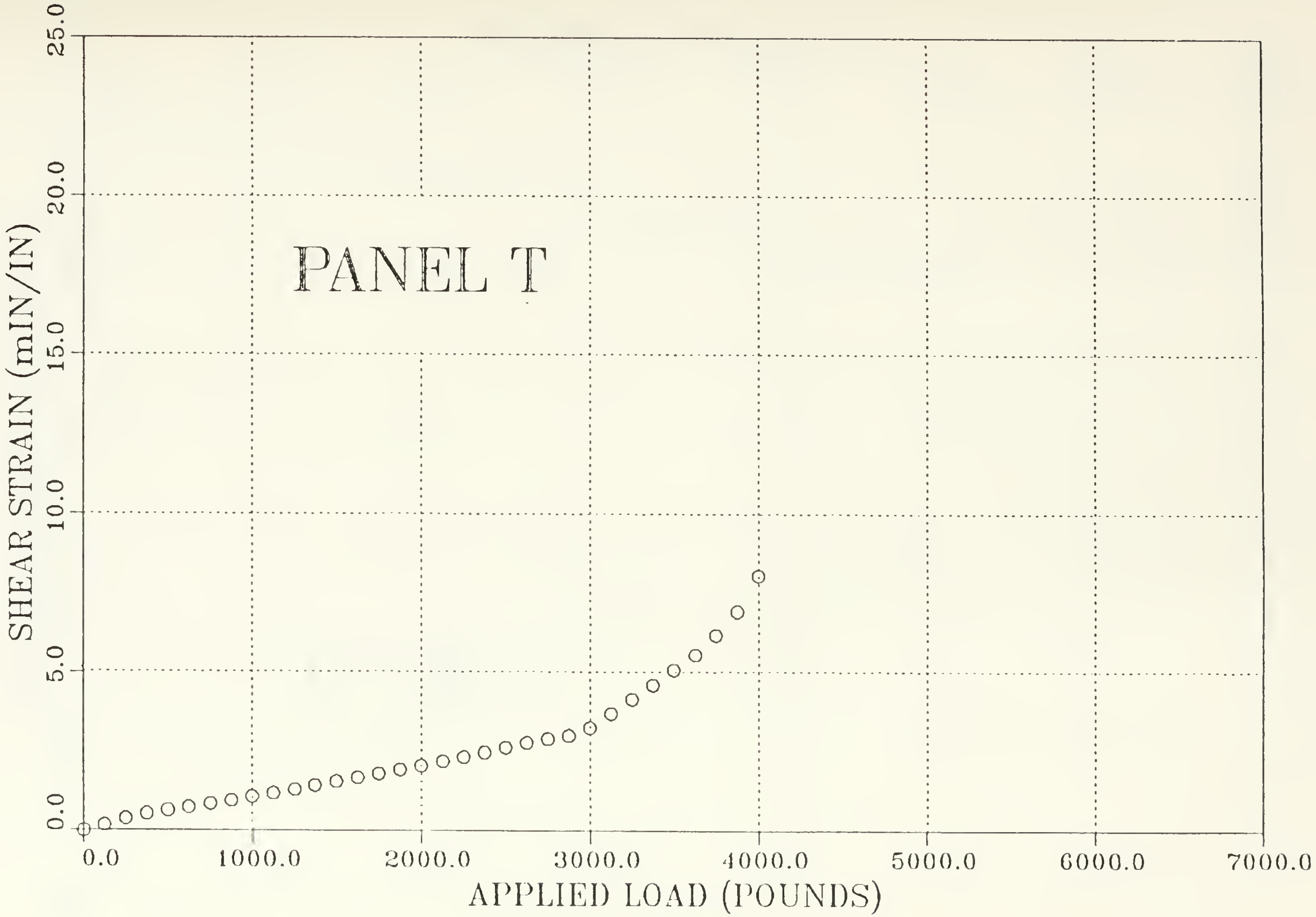
Panel Q - 1/2" Hole, Kevlar Patch, Foam Plug, 1/4" Overlap



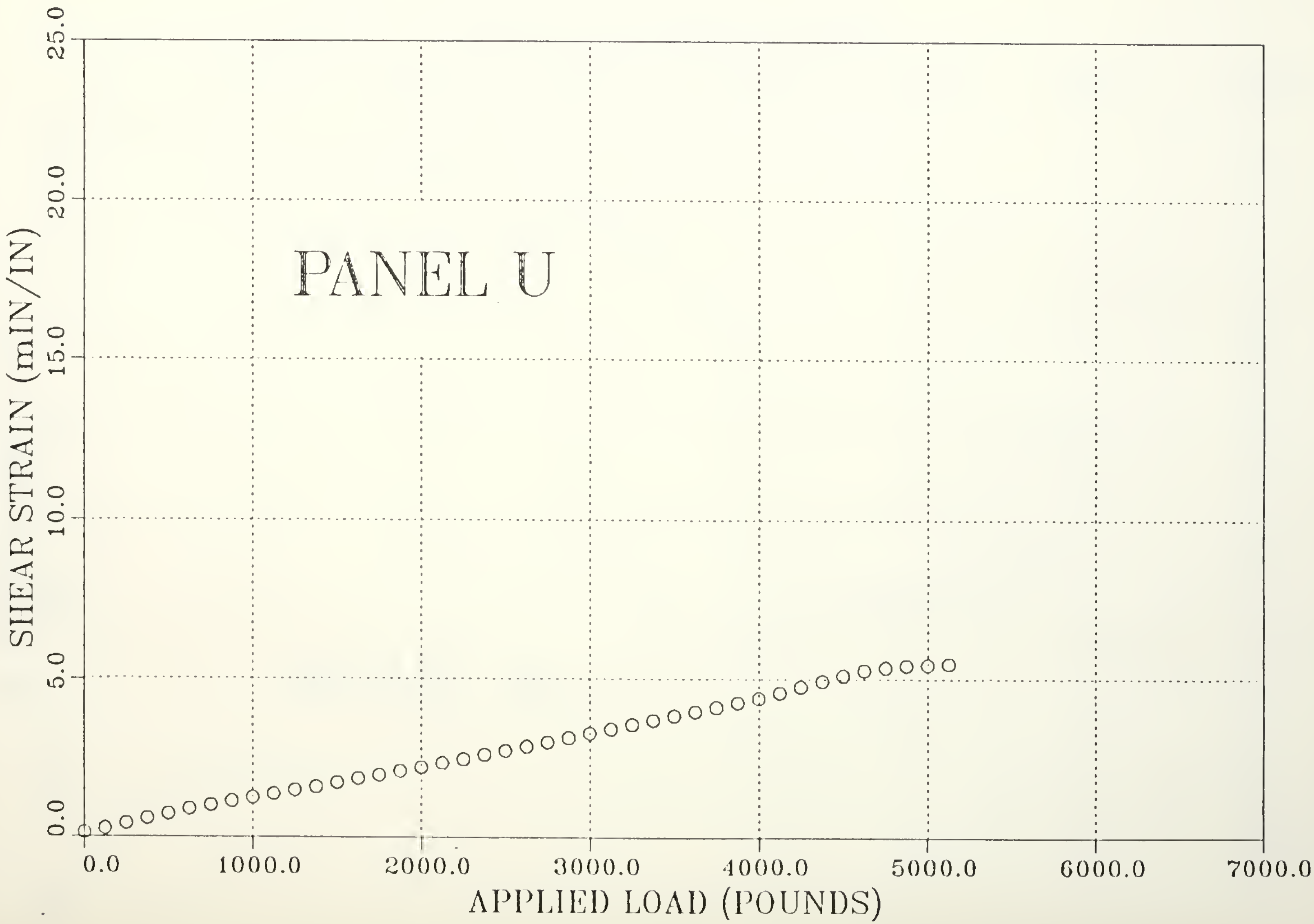
Panel R - 1/2" Hole, Kevlar Patch, Foam Plug, 1/2" Overlap



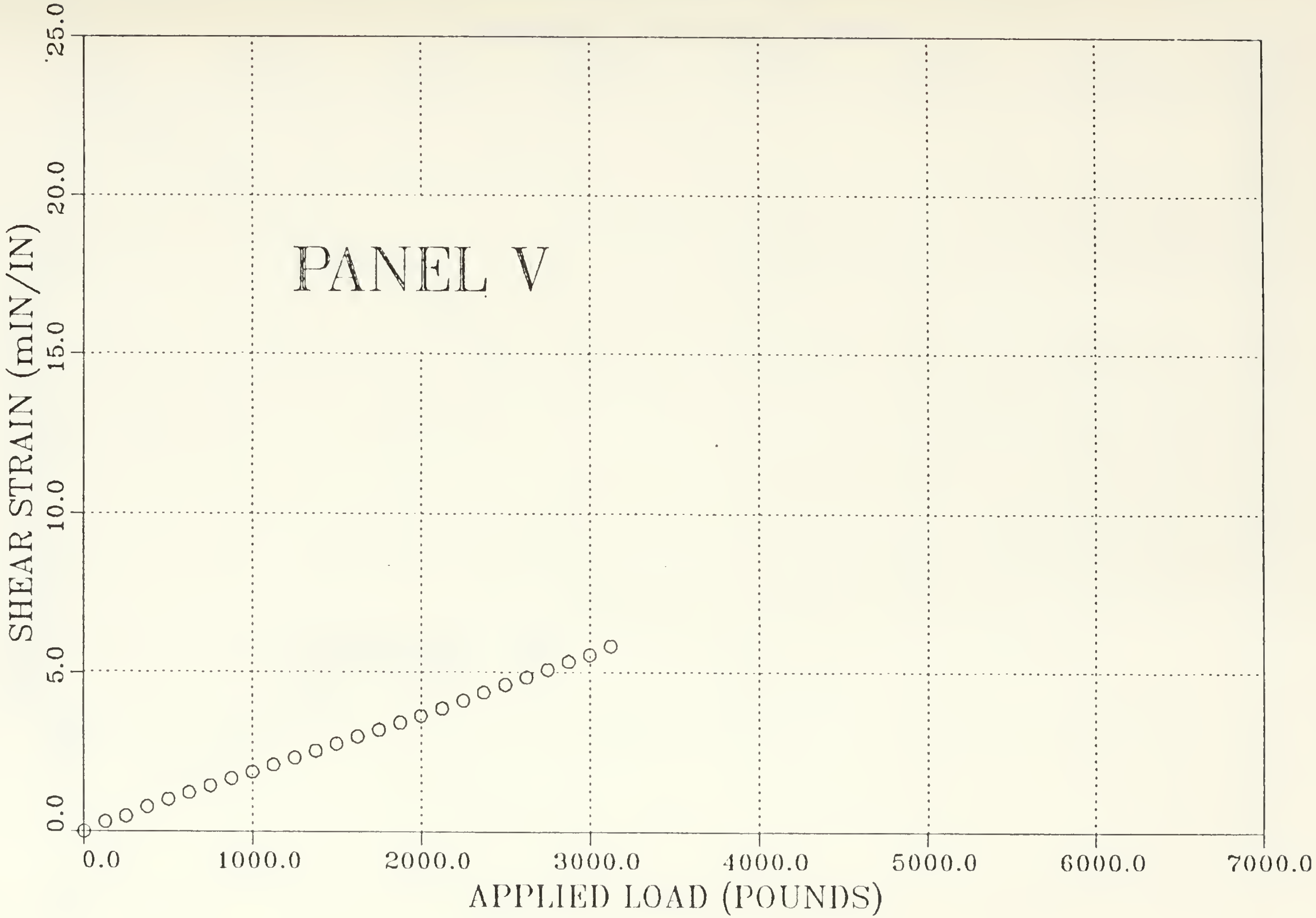
Panel S - 1/2" Hole, No Patch



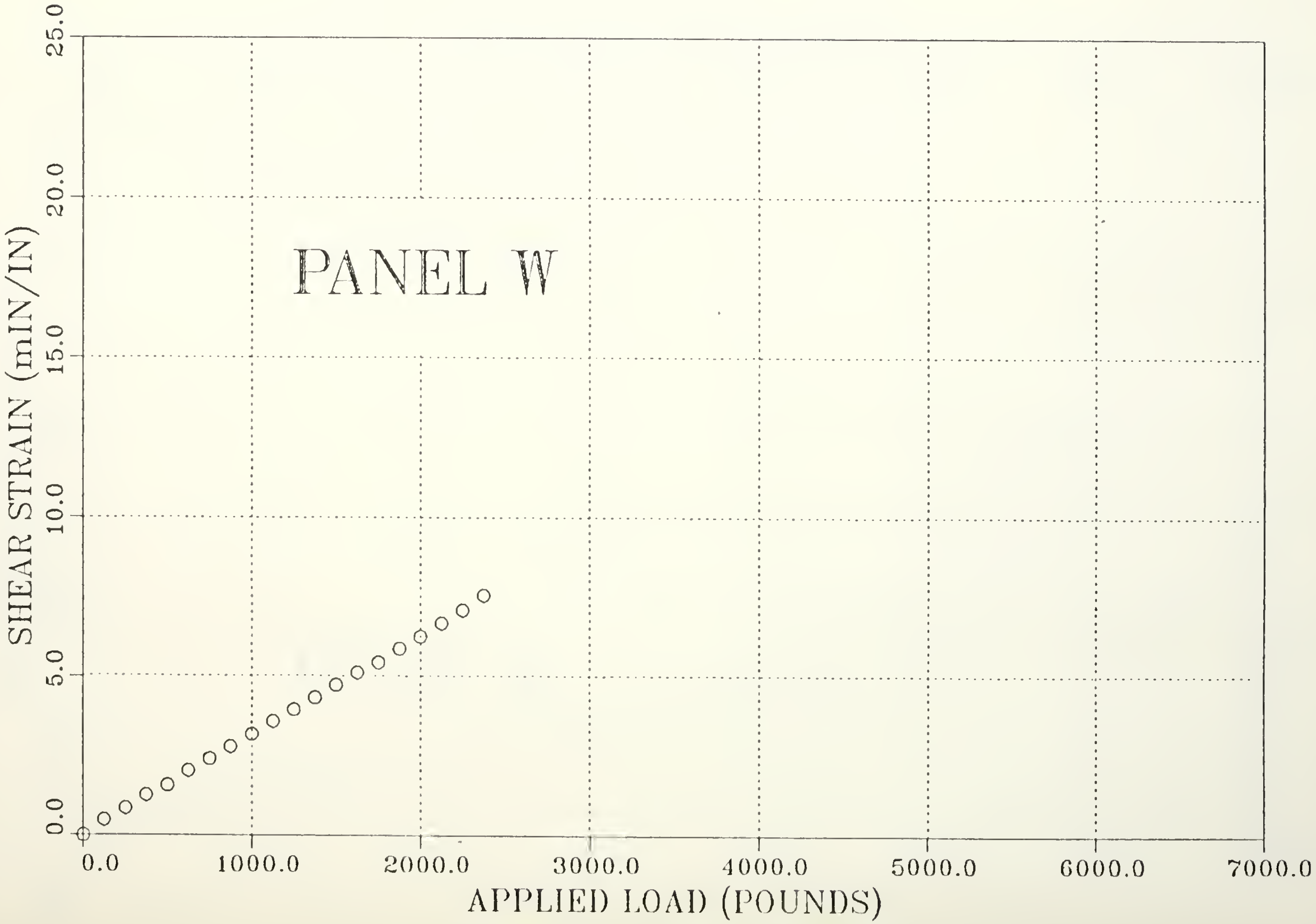
Panel T - 2" Hole, No Patch



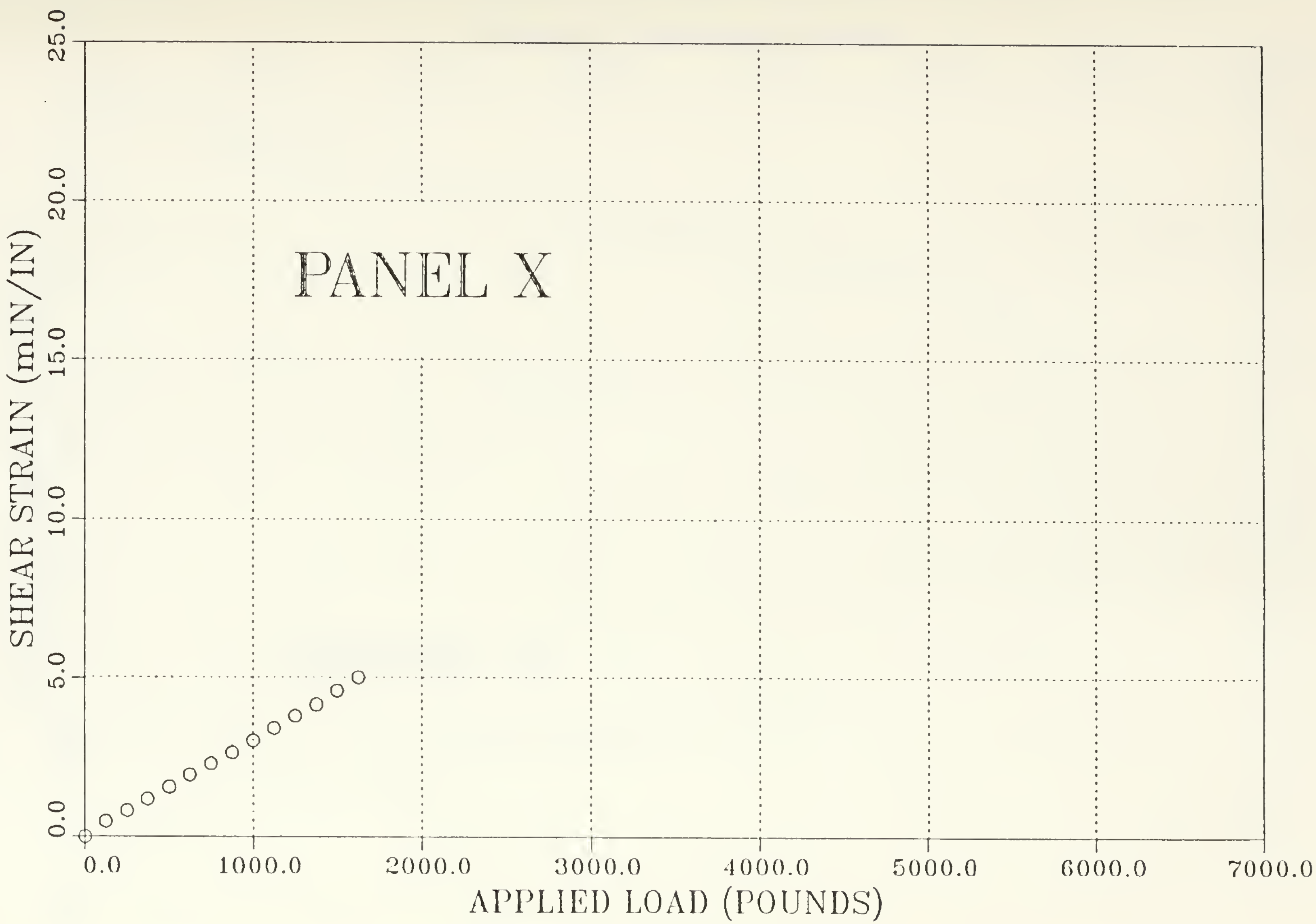
Panel U - 1" Hole, No Patch



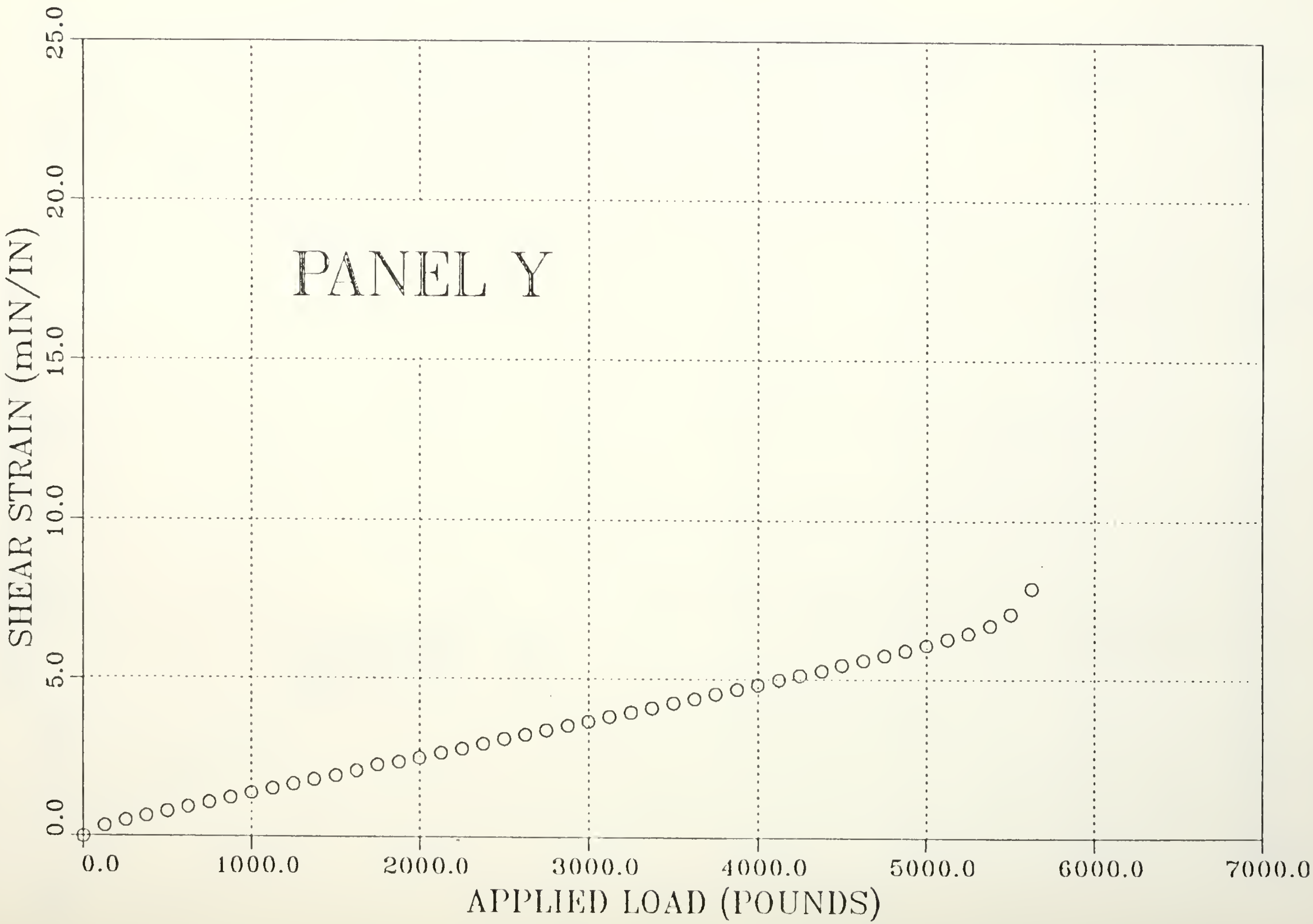
Panel V - 3" Hole, No Patch



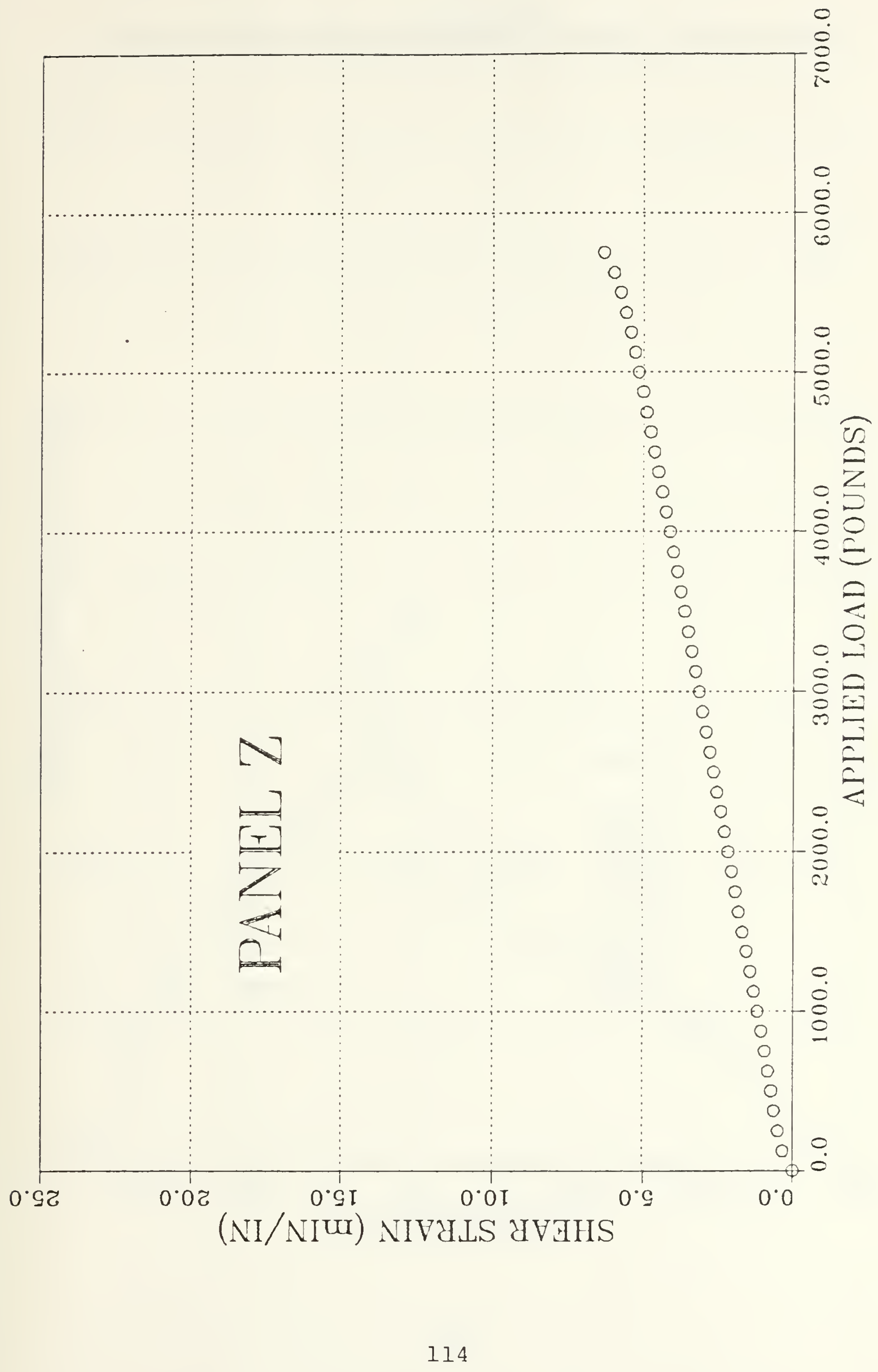
Panel W - 4" Hole, No Patch



Panel X - 5" Hole, No Patch



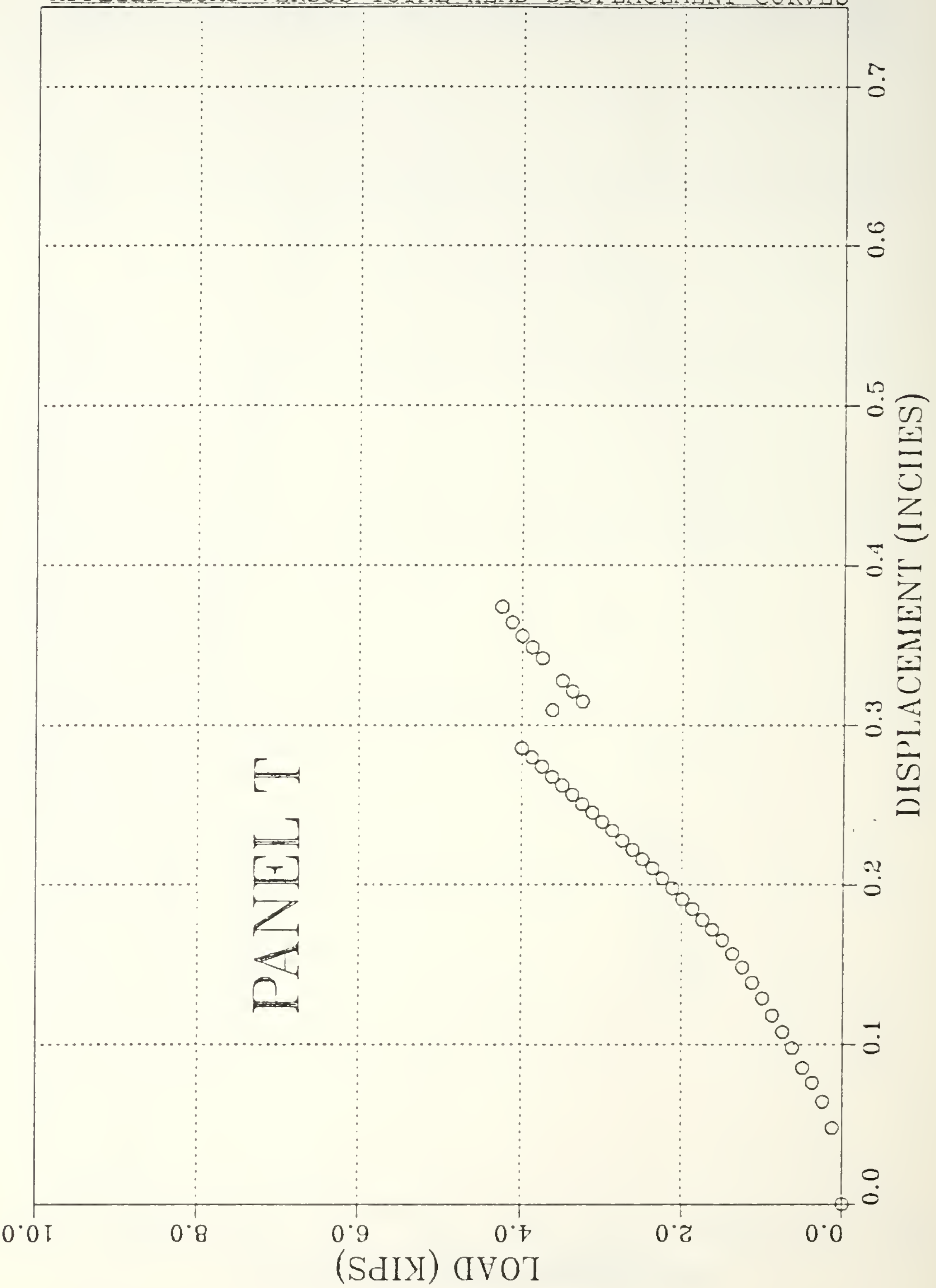
Panel Y - Solid Panel



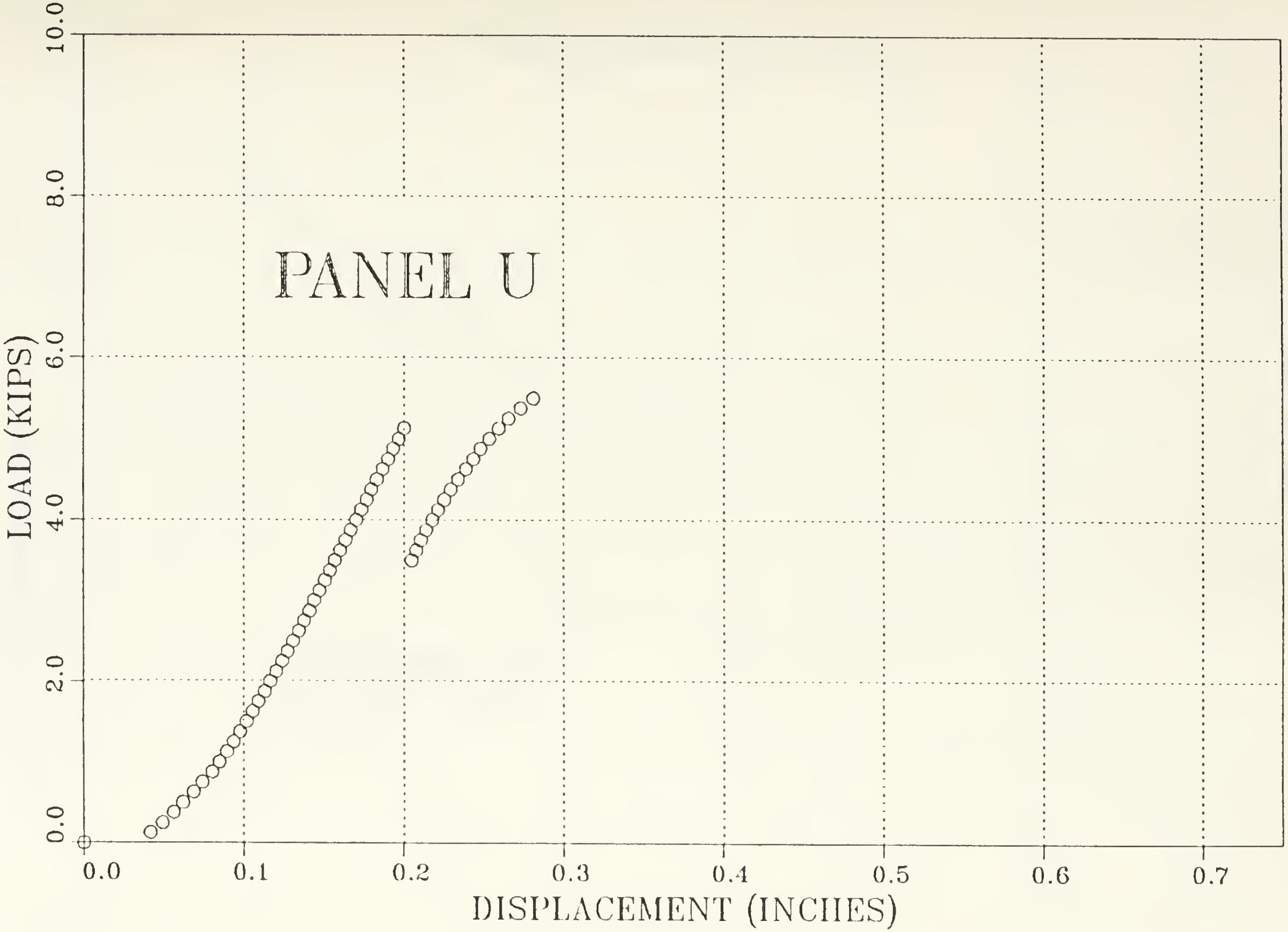
Panel Z - Solid Panel

APPENDIX C

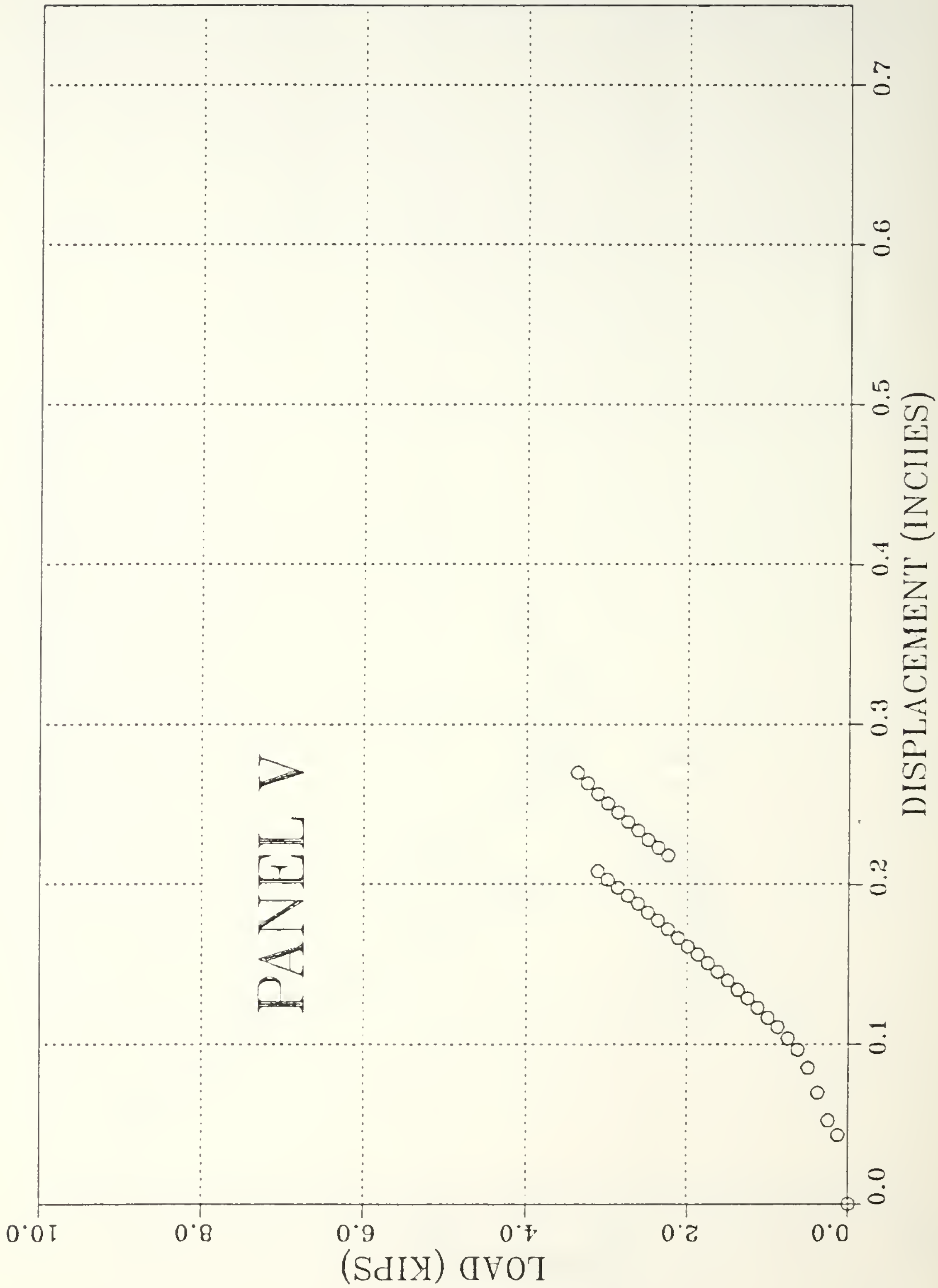
APPLIED LOAD VERSUS TOTAL HEAD DISPLACEMENT CURVES



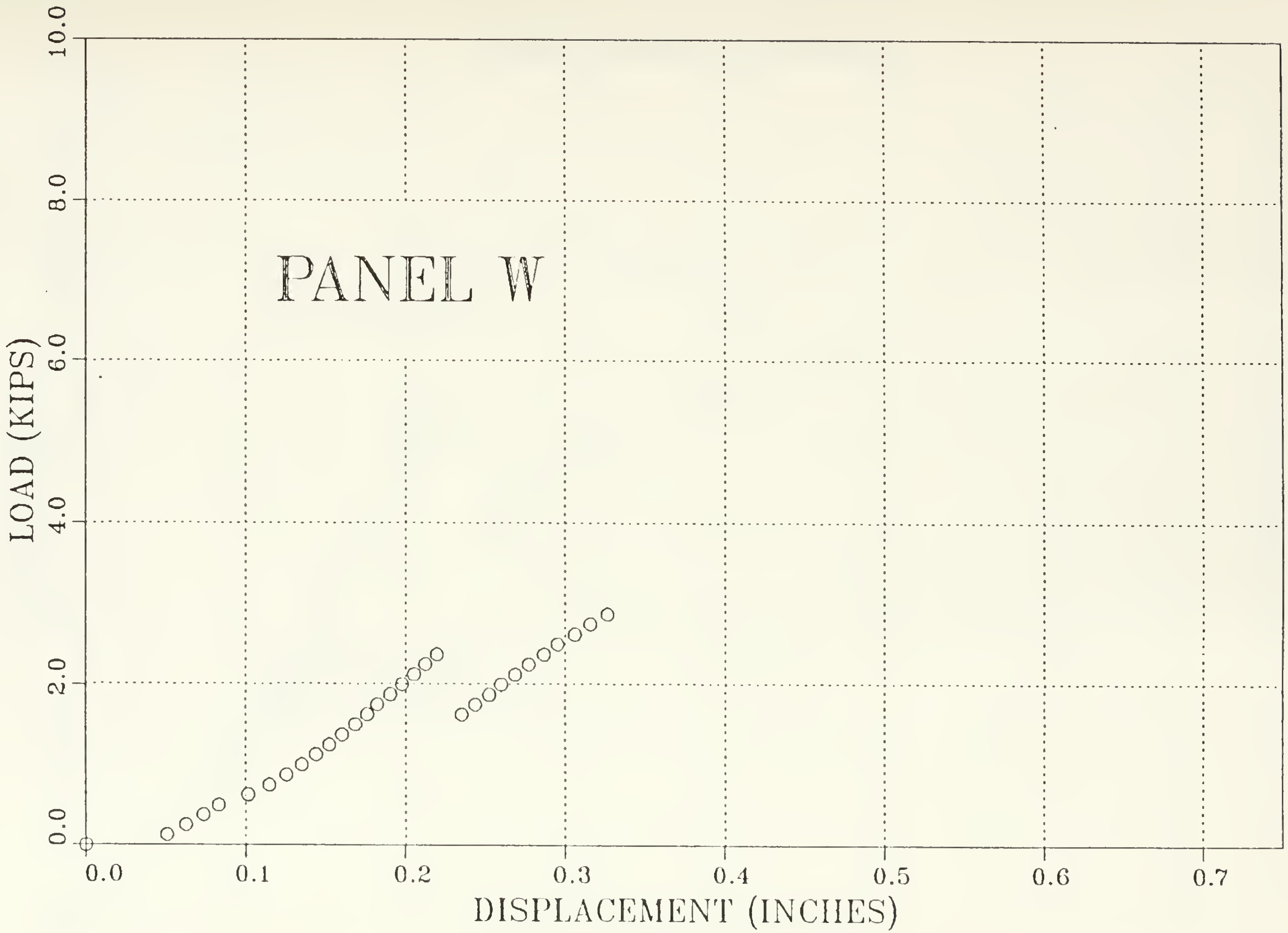
Panel T - 2" Hole, No Patch



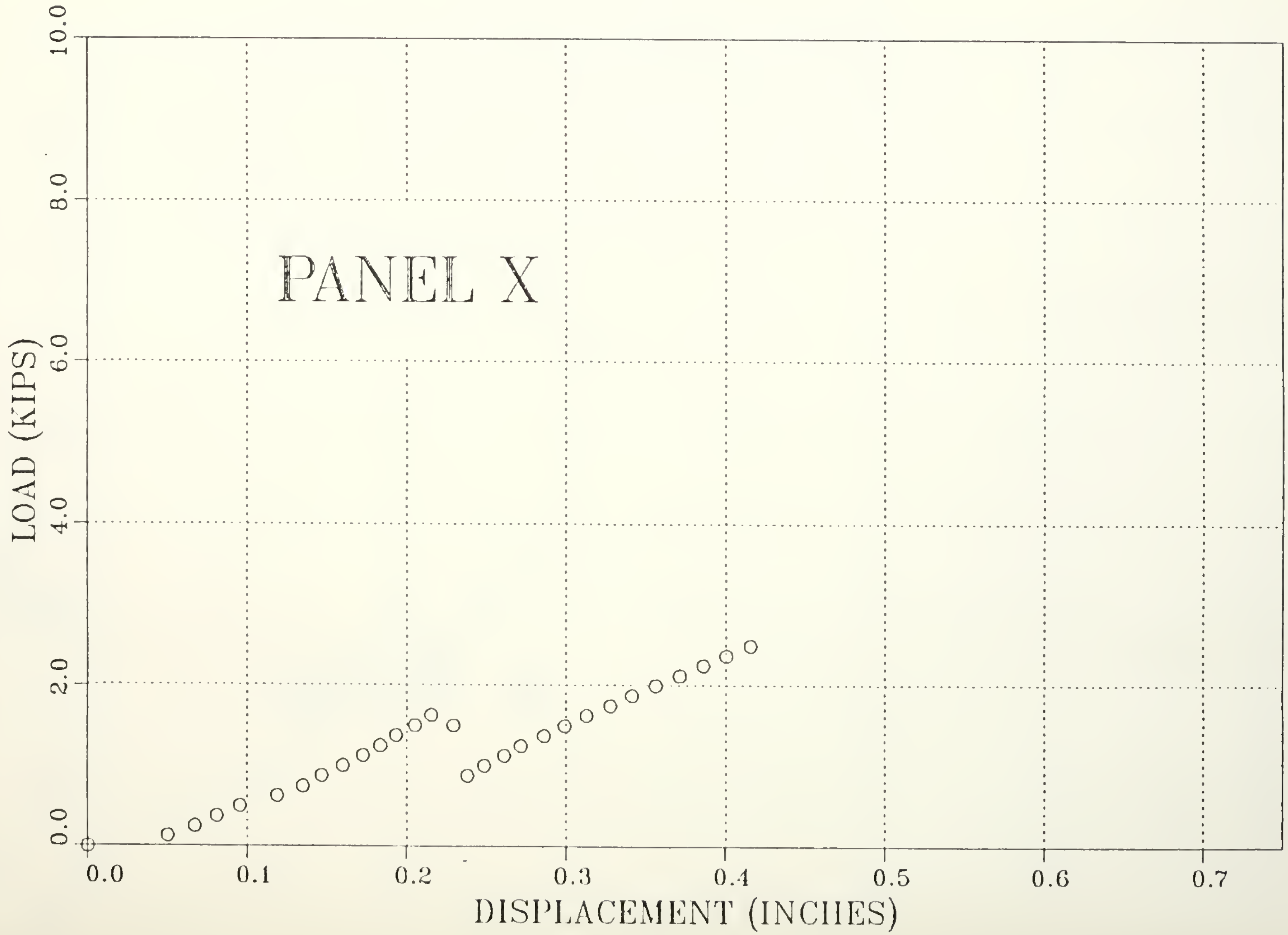
Panel U - 1" Hole, No Patch



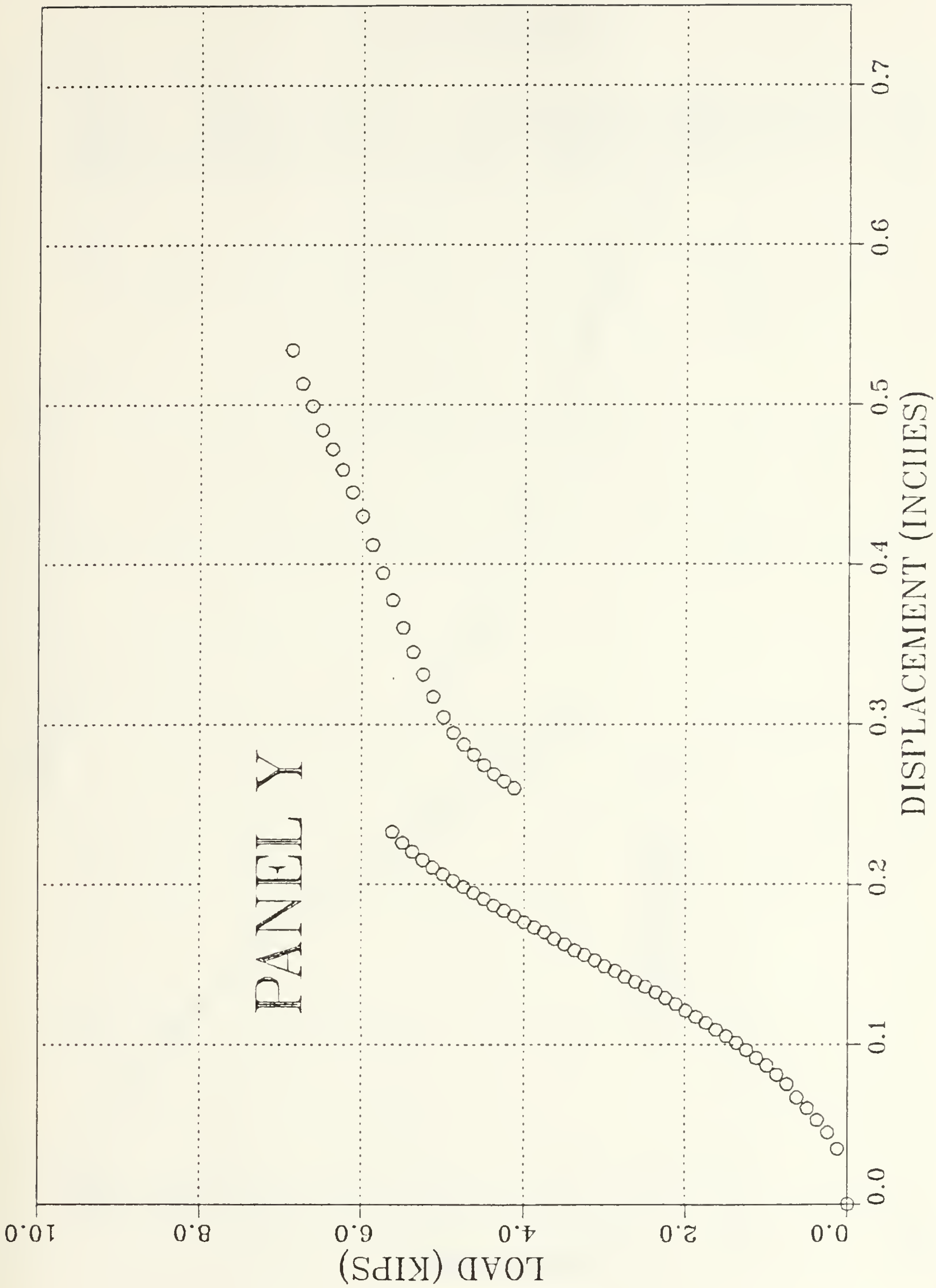
Panel V - 3" Hole, No Patch



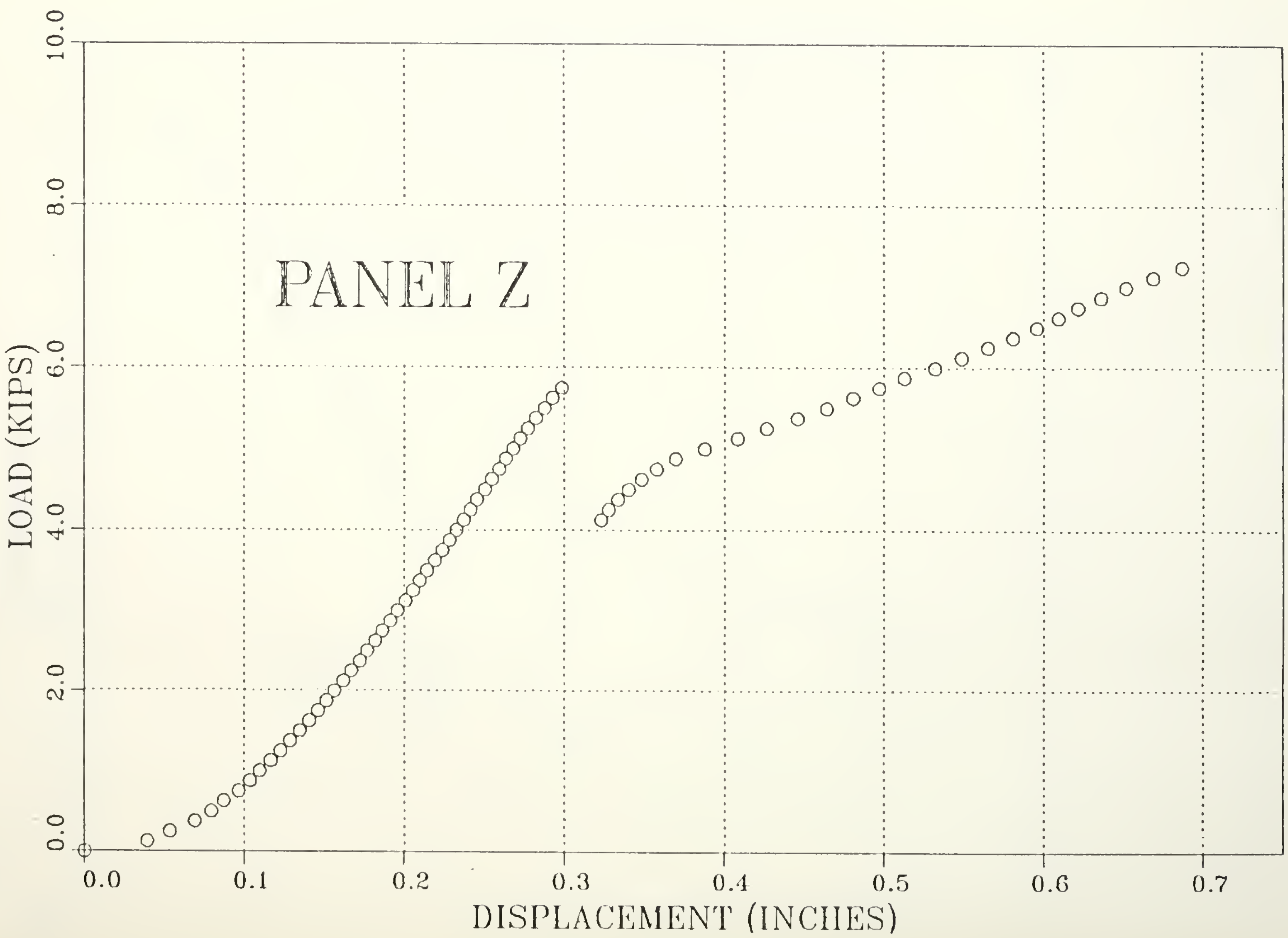
Panel W - 4" Hole, No Patch



Panel X - 5" Hole, No Patch



Panel Y - Solid Panel



Panel Z - Solid Panel

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Farley, G.L. and Baker, D.J., In-Plane Shear Test of Thin Panels, presented at the 38th Annual Forum of the American Helicopter Society, Anaheim, California, May 1982.

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